NASA CR 132362

COMPUTER PROGRAM TO PERFORM COST AND WEIGHT ANALYSIS OF TRANSPORT AIRCRAFT

FINAL REPORT

VOLUME II ✦ TECHNICAL VOLUME

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GENERAL DYNAMICS
Convair Aerospace Division

COMPUTER PROGRAM TO PERFORM COST AND WEIGHT ANALYSIS OF TRANSPORT AIRCRAFT

VOLUME I ✦ SUMMARY
VOLUME II ✦ TECHNICAL VOLUME

I

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FINAL REPORT

VOLUME II ✦ TECHNICAL VOLUME

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SUMMARY

This report presents the results of a research and development study performed under NASA contract NAS 1-11343. The objective of this study was to develop an improved method for estimating aircraft weight and cost using a unique and fundamental approach originated by Convair Aerospace. The results of this study were integrated into a comprehensive digital computer program, which is intended for use at the preliminary design stage of aircraft development. The program provides a means of computing absolute values for weight and cost, and enables the user to perform trade studies with a sensitivity to detail design and overall structural arrangement. Both batch and interactive graphics modes of program operation are available. The report is documented in three volumes – the Final Report Summary, the Final Report Technical Volume, and the User's Manual.

The cost derivation portion of the program encompasses the areas of manufacturing and material cost, engineering cost, tooling cost, total vehicle program cost, and a return-on-investment analysis. The approach provides an accounting of aircraft weight and cost elements beginning with initial conceptual design studies and continuing through detail design, aircraft production, and flight operations. The fundamental weight and cost driver is the definition of a detail parts listing that is generated for a given vehicle when only conceptual details of a configuration are available for use as input. The detail parts from this listing are analyzed individually to determine their weight and cost. Summations are made, adding in weight and costs of assembly elements, to determine the complete vehicle weight and manufacturing cost.

The detail part level breakdown of components is attained through the use of several synthesis routines coupled in a series. A vehicle synthesis routine acts as the overall driver. The input consists of generalized vehicle and mission parameters that are typically known at the preliminary design level. This routine sizes the overall vehicle, generates major vehicle component weights, and derives a large amount of overall vehicle geometry. The output from this routine is used, in turn, to drive a structural synthesis routine that sizes, weighs, and derives geometry for major subcomponents. The detail part definition process follows, which calls out, for each of the major subcomponents specified, a list of the typical detail parts making up the subcomponent. These detail parts then represent the basis of the fundamental level for the weight and cost analysis.

The computer program is written in Fortran IV and is designed for use on CDC 6000 series computers. Several test cases, using data for existing aircraft, were run to check the program results against actual data. It was shown that

the program represents an accurate and useful tool for estimating purposes at the preliminary design state of airframe development. A sample case along with an explanation of program applications and input preparation is presented in the User's Manual volume of this report. Table 1 is a summary of the program functional capability and Figure 1 is a program block diagram.

Table 1. Summary of the Program Functional Capability

Vehicle Synthesis (Sizing)	Manufacturing Cost
Aircraft Balance	Material Cost
Mission Center of Gravity Envelope	Engineering Cost
Area Ruled Fuselage Geometry	Tooling Cost
General Curve Plotting	Total Vehicle Program Cost
Structural Synthesis	Return-on-Investment
Parts Definition and Weight	Airline Route Analysis

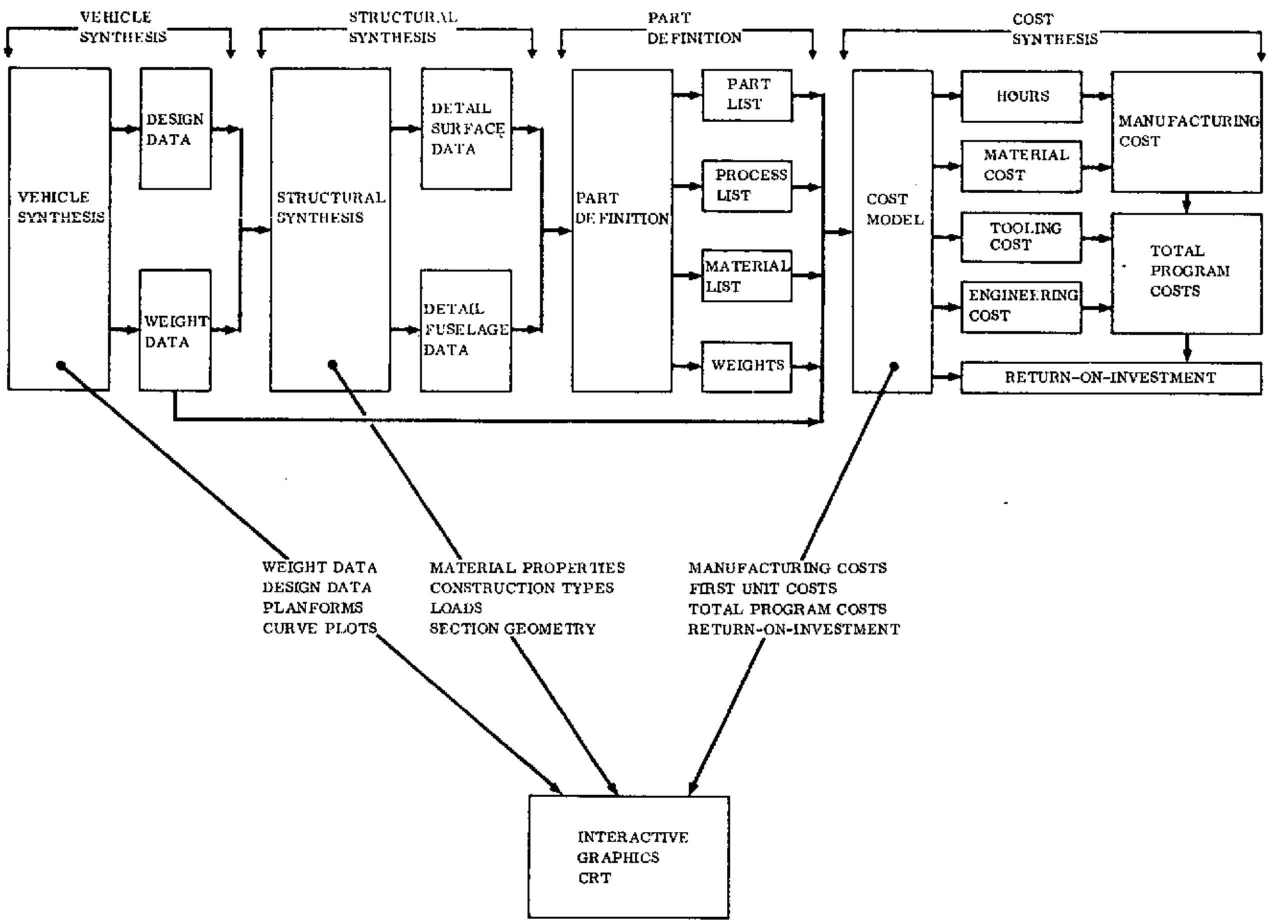


Figure 1. Vehicle Design and Evaluation Program (VDEP) Block Diagram

SECTION 1

INTRODUCTION

With the steadily rising cost of aircraft production and operation, and with the large number of materials and structural design concepts applicable to flight vehicles, it becomes increasingly important to be able to assess the impact of aircraft design alternatives in terms of cost and performance. A major deficiency of past cost-estimating methods has been the result of an over-reliance on vehicle weight as a cost driving variable. Assuming the use of conventional materials and structural methods, weight was indeed a useful parameter in cost studies. However, advances in technology have produced components of increased specific strength, and hence, decreased weight, but at the expense of requiring increasingly exotic materials and fabrication complexities. The result has been an inverse in the typical cost/weight relationship. A second deficiency of previous cost-estimating methods has been the use of oversimplified cost models that lack the depth necessary to provide a sensitivity to design tradeoff choices in terms of structural materials and methodology.

The objective of this study was to develop a digital computer program for evaluating the weight and costs of advanced transport designs. The resultant program, intended for use at the preliminary design level, incorporates both batch mode and interactive graphics run capability. The basis of the weight and cost estimation method developed is a unique way of predicting the physical design of each detail part of a vehicle structure at a time when only configuration concept drawings are available. In addition, the technique relies on methods developed at the San Diego Operation to predict the precise manufacturing processes and the associated material required to produce each detail part.

The starting point of the present effort was a computer program developed under NASA Contract NAS 2-5718, Estimation of Airframe Manufacturing Costs (Reference 1). The previous study was fundamental in establishing the feasibility of the methodology to be applied. Incompassing the areas of manufacturing and material cost, engineering cost, tooling cost, total vehicle program cost, and return-on-investment, the current study represents a significant extension and refinement of the methods originally formulated. Preliminary weight and cost estimates for transport aircraft composite structural design concept (Reference 17) was also utilized within this study effort.

Weight data are generated in four areas of the program. Overall vehicle system weights are derived on a statistical basis as part of the vehicle sizing process. Theoretical weights, actual weights, and the weight of the raw material to be purchased are derived as part of the structural synthesis and part definition processes based on the computed part geometry.

The manufacturing cost analysis, based at the individual detail part level, is made by considering the actual manufacturing operations reuired to produce that part. A list of shop operations is called out with each detail part, and a series of equations associated with each operation is used to compute the shop hours necessary to make the part.

By applying the appropriate labor rates to the calculated hours, the direct and indirect manufacturing labor costs are found. Material costs are computed based on the amount of material required to manufacture each part.

Engineering costs are computed based on the number of manhours necessary to perform the various tasks associated with the development and production of aircraft. The computation has as its basis equations originally developed by Levenson and Barro of the Rand Corporation. Initial engineering hours are broken down and distributed among the various engineering disciplines based on studies made of historical data.

Tooling costs are computed as a function of the number of basic tool manufacturing hours, initial and sustaining aircraft production rates, and tooling labor rates. Basic tool manufacturing hours are derived as a function of the number of dissimilar parts to be produced, the average number of tools required per dissimilar part, and the average number of hours required to produce each tool.

Total vehicle program costs are computed based on a cost model that was assembled primarily from the work of Kenyon. Cost elements that are computed elsewhere in the program are brought across and substituted into the model. A learning-curve approach is utilized to derive costs of a given unit or lot as a function of the first unit cost.

A comprehensive measure of the total economic viability for a commercial transport operation is reflected in the return-on-investment analysis. Direct operating costs are computed using the 1967 Air Transport Association formula updated to 1972 cost levels. Indirect operating costs and return-on-investment are computed by applying aircraft acquisition and direct operating costs to a defined traffic structure. Output includes direct operating costs, indirect operating costs, revenue, load factors, profit, return-on-investment, and fleet size.

One advantage provided by the method developed is its capability to make trade studies from several levels of consideration. For example, weight and cost data can be related directly to key system parameters at the vehicle mission level such as payload, speed, range, and landing field length requirements. At the vehicle configuration level, data can be related directly to surface areas, span, sweep, taper, etc., and fuselage length, slenderness, etc. At the major component level comparisons can be made between different materials, modes of construction, detail part make-up, etc. Tradeoffs can be made to determine the overall vehicle weight and cost sensitivities at each of these levels, and in this manner the proposed aircraft design may be further and further refined down to high degree of detail. Thus, engineering functions can gain insight into the cost effectiveness of alternate aircraft systems, perform design trade studies, and perform studies to determine the impact of more detailed engineering alternatives with respect to a particular aspect of a design.

A second advantage of the method is its overall accuracy in estimating weight and cost. The increased accuracy is derived from the fundamental level of the analysis technique, starting from the detail part level. Each part and each assembly is accounted for to establish its individual effect on the cumulative total.

SECTION 2

TECHNICAL DISCUSSION

The essential features exhibited by the resultant weight and cost analysis program can be categorized into three major areas: the vehicle synthesis, structural synthesis, and cost synthesis. The vehicle synthesis provides overall vehicle size, balance, and dimensional data. The structural synthesis provides component and subsystem sizing, and part definition data. The cost synthesis provides manufacturing, material, engineering, tooling, and total vehicle program costs, and a return-on-investment analysis.

The vehicle synthesis process provides a rapid means of initially sizing the vehicle to derive overall vehicle geometry, weight, and balance. Input requirements are at a level typically available at the preliminary design stage of aircraft development. Output, which acts as a driver for the structural synthesis routines, is comprised of a group weight statement, group cg statement, vehicle geometry data, loads data, wing station data, engine data, landing gear data, and a mission cg range display. A detailed technical discussion, including the equations utilized within the vehicle synthesis routines, is presented in Section 2. 1.

The structural synthesis process provides detailed geometry, loads, and weight data for the primary structural elements associated with the aerodynamic surfaces and the basic fuselage structural shell. The synthesis utilizes a multistation analysis approach that assumes a reasonable structural continuity and a well defined elastic axis. The aerodynamic surface leading edge, trailing edge, and tips, the fuselage penalty items, and the detail part breakdown of the major surface and body components are accounted for in associated part definition routines. The structural synthesis provides the driving parameters for the part definition routines, which in turn provide the driving parameters for the cost synthesis. A detailed technical discussion of the strucutral synthesis and the part definition portions of the program is presented in Section 2. 2.

The cost analysis portion of the program provides: manufacturing costs based on a consideration of the actual detail parts to be produced and the actual manufacturing and assembly processes required to produce them; material costs based on the type and quantity of material actually purchased; engineering costs based on a statistical treatment of historical data; tooling costs based on the number of parts to be produced; total vehicle program costs based on a cost estimating relationship (CER) approach; and a return-on-investment analysis. Except for the total vehicle program cost and the return-on-investment analysis, input to the cost portion of the program is primarily self generated, comprised of either values that have been derived by the preceding synthesis routines or values that are generated internally as needed. A capability has been designed into the program to allow direct input of any parameters for which values are known or for which a constant value is desired. Input to the total vehicle program cost routine is comprised of a series of CER's that are typical of that particular type of analysis. A detailed discussion of the cost computations is presented in Section 2. 3.

2.1 VEHICLE SYNTHESIS

The purpose of the vehicle synthesis process is to provide a means for the preliminary design analyst to initially define the size and weight characteristics of a projected vehicle and its basic components. At the conceptual design stage only generalized mission and performance requirements are known, and hence available as input for the initial vehicle sizing studies. These relatively few parameters are used in the synthesis process to generate more detailed vehicle performance, weight, and geometry data. While the required input is kept at a minimum, many parameters are defined as optional input and are normally generated internally. As more detailed vehicle data is defined and fixed, the optional data may be input directly to override the internally generated data. In this way the process remains useful throughout the design definition and the fixed-configuration refinement stages of study.

An existing vehicle sizing program (documented in References 2 and 3) was expanded in scope and modified for use with interactive computer graphics. The basic routines encompassed by the vehicle synthesis process include the following: geometry, weight, performance, balance, area distribution, and cg range. The geometry subroutine derives geometry data for the fuselage, wing, horizontal and vertical stabilizers, landing gear, and engines. Basic lengths, widths, depths, areas, wetted areas, and volumes are included. The result is sufficient data to allow the construction of a general arrangement three-view drawing of the sized vehicle. Input to the geometry subroutine is made directly, and, at the user's option, may include parameters generated by the performance subroutine. Output is used to drive the weight and area distribution subroutines.

The weight subroutine weighs the sized vehicle using statistically based weight equations. A value for initial design weight is input to establish a starting point for the sizing process. The design weight is subsequently readjusted as required to satisfy specified mission and performance requirements. The output is assembled in the format of the group weight statement defined by MIL-STD-254(ASG).

The performance subroutine provides a simplified analysis accounting for vehicle performance, propulsions, and loads. The method is intended to supply reasonable input data only in a limited number of applications, and is not designed to replace the more general analyses found in larger, more comprehensive programs. Input is comprised of parameters such as landing field length, takeoff field length, climb requirements, fuel requirements, cruise speed, range, wing loading, and aspect ratio. Output includes: lift and drag coefficients, wing area, thrust-to-weight ratios, fuel requirements, and ultimate gust load factor.

The balance subroutine utilizes the output from the weight analysis plus generalized location coefficients to compute the cg location for each aircraft item listed on the group weight statement. Vehicle cg locations are output as both a station and as a percentage of the wing mean aerodynamic chord for several aircraft weight conditions.

The area distribution subroutines use geometry data to generate an area ruled fuselage shape for assumed Mach 1.0 conditions. The user may view the resultant vehicle on the graphics console, and reshape the fuselage to satisfaction. The approach allows the selection of an idealized curve typical of a vehicle of the type under consideration, and the specification of maximum and minimum fuselage area constraints.

The cg range subroutines provide the capability to plot the cg envelope for a given vehicle "flying" a defined mission profile. Input is comprised of weight and cg data previously derived. The output allows the user to view the cg travel as various aircraft expendables are loaded onto the aircraft or are used.

The actual sizing process within the vehicle synthesis is driven by an iteration process. A test is made to see whether or not the final vehicle gross takeoff weight is within set limits (±0.1%) with respect to the initial design weight. This test ensures that the final sized vehicle computed weight is consistent with specified mission and performance requirements, and consistent with the weight values used in calculations in previous subroutines. The following logic applies:

Test: is $\dfrac{WT_{Initial} - WT_{Calc.}}{WT_{Calc.}} < 0.1\%$

If yes: Display weight statement

If no: Set $WT_{Initial} = WT_{Calc.}$ and reiterate the sizing process

The initial value for $WT_{Initial}$ is supplied by the user. Each of these routines is described in detail in following sections. A dictionary of terms from the vehicle synthesis routines is presented in Table 2-1. All of the required computer program input in the area of the vehicle synthesis utilizes the British system of units.

Table 2-1. Definition of Terms – Vehicle Synthesis

a	=	Landing deceleration
$A_{Body(max)}$	=	Maximum cross-sectional area of body
$A_{c.s.}$	=	Cross-sectional area of total surface at any spanwise location
$A_{c.s. Box}$	=	Cross-sectional area of surface structural box at any spanwise location
$A_{c.s. Box 1}$	=	Cross-sectional area of surface structural box at spanwise location No. 1

Table 2-1. Definition of Terms – Vehicle Synthesis, Contd

$A_{c.s. Box 2}$	=	Cross-sectional area of surface structural box at spanwise location No. 2
$A_{c.s.(Nacelle)}$	=	Maximum cross-sectional area of engine nacelle
ALT	=	Altitude
A_{max}	=	Maximum cross-sectional area of total airplane
AR	=	Wing aspect ratio
AR_H	=	Horizontal stabilizer aspect ratio
AR_V	=	Vertical fin aspect ratio
b	=	Wing span
B	=	Maximum outside body width along wing chord
b_H	=	Horizontal stabilizer span
BM	=	Approximation of root bending moment
C_1	=	Temperature coefficient
C_2	=	Wing relieving load coefficient
C_B	=	Wing chord at the body
C_{Box}	=	Wing chord of the structural box
C_{Brk}	=	Wing chord at the wing break
C_{Dcr}	=	Coefficient of drag — cruise
C_{Df}	=	Coefficient of skin friction
C_{Do}	=	Coefficient of drag — equivalent profile

Table 2-1. Definition of Terms – Vehicle Synthesis, Contd

C_f	=	Flap chord
CG_{bac}	=	Summation of cg station locations of all aircraft items classified under body and contents
$CG_{.25\bar{c}}$	=	Wing quarter-chord station location
$CG_{.\bar{c}w}$	=	Total aircraft cg location expressed in percent wing MAC
CG_T	=	Total aircraft cg station location (station zero is considered the aircraft nose)
CG_{wac}	=	Summation of cg locations (relative to wing quarter chord) of all aircraft items classified under wing and contents
C_{Hvol}	=	Horizontal tail volume coefficient
$(C_L/C_D)_{cr}$	=	Lift-to-drag ratio — cruise [(L/D)cr]
C_{Lcr}	=	Coefficient of lift — cruise
C_{Lland}	=	Coefficient of lift — landing
$(C_L/C_D)_{max}$	=	Maximum lift-to-drag ratio [(L/D)max]
$C_{L_{max}}$	=	Maximum lift coefficient
$C_{L_{TO}}$	=	Coefficient of lift at takeoff
C_N	=	Wing chord at any spanwise location
C_{N_h}	=	Horizontal tail normal force coefficient
$C_{N_{(maneuver)}}$	=	Vertical tail normal force coefficient
C_{N_β}	=	Vertical tail normal force curve slope
$C_{N_{\alpha h}}$	=	Horizontal tail normal force curve slope
$C_{N\alpha w}$	=	Wing normal force curve slope
C_r	=	Wing chord at the root

$C_{r(extended)}$	=	Wing chord at the extended root
C_{SP}	=	Spoiler chord
C_t	=	Wing chord at the tip
$C_{V_{vol}}$	=	Vertical tail volume coefficient
$\bar{C}_w$	=	Wing mean aerodynamic chord
D_{Body}	=	Diameter of body
D_{cl}	=	Climb distance
D_{engine}	=	Diameter of engine
$D_{nacelle}$	=	Diameter of nacelle
e	=	Wing efficiency factor
F_a	=	Allowable bending stress at wing root
$F_{D_{main}}$	=	Drag load on main landing gear
$F_{D_{nose}}$	=	Drag load on nose landing gear
$F_{H_{(gust)}}$	=	Gust load on horizontal tail
$F_{H(maneuver)}$	=	Maneuver load on horizontal tail
$F_{H(ult)}$	=	Ultimate horizontal tail load
$F.R._{B(equiv)}$	=	Equivalent fineness ratio of body
$F_{V(gust)}$	=	Gust load on vertical tail

Table 2-1. Definition of Terms – Vehicle Synthesis, Contd

F_{Vmain}	=	Vertical load on main landing gear
$F_{V(maneuver)}$	=	Maneuver load on vertical tail
F_{Vnose}	=	Vertical load on nose landing gear
$F_{V(ult)}$	=	Ultimate vertical tail load
g	=	Gravitational acceleration
GOH_{min}	=	Minimum ground object height
H	=	Height of body
K	=	Gust response factor
$k_{C_{Lmax}}$	=	Maximum lift coefficient scaling factor
K_{Des}	=	Design weight scaling factor for landing gear weight
k_{e_1}	=	Engine weight ratio scaling factor
k_{e_2}	=	Engine weight ratio scaling factor
k_{GE}	=	Ground effect factor on wing loading at landing
k_{land}	=	Landing weight scaling factor for landing gear weight
k_m	=	Maximum design weight scaling factor for landing gear weight
k_1	=	Landing design weight scaling factor
k_2	=	Percent mission fuel for landing design weight
k_3	=	Percent mission fuel for design mission weight
k_4	=	Maximum design weight scaling factor

Table 2-1. Definition of Terms – Vehicle Synthesis, Contd

k_5	=	Body length scaling factor
k_6	=	Body wetted area scaling factor
k_7	=	Main oleo length scaling factor
k_8	=	Engine nacelle diameter scaling factor
k_9	=	Engine nacelle length scaling factor
k_{10}	=	Engine nacelle wetted area scaling factor
k_{11}	=	Allowable bending stress at wing root scaling factor
k_{12}	=	Wing relieving load scaling factor
k_{13}	=	Flap weight scaling factor
k_{14}	=	Leading edge device weight scaling factor
k_{15}	=	Spoiler weight scaling factor
k_{16}	=	Wing fold penalty weight scaling factor
k_{17}	=	Wing weight scaling factor
k_{18}	=	Horizontal tail weight scaling factor
k_{19}	=	Vertical tail weight scaling factor
k_{20}	=	Fuselage weight scaling factor
k_{21}	=	Surface control weight scaling factor
k_{22}	=	Surface control weight scaling factor - alternate method
k_{23}	=	Inlet, cooling, lubrication and starting system weight scaling factor
k_{24}	=	Engine control weight scaling factor
k_{25}	=	Fuel distribution system weight scaling factor – function of number of tanks

Table 2-1. Definition of Terms – Vehicle Synthesis, Contd

k_{26}	=	Fuel distribution system weight scaling factor – function of fuel quantity
k_{27}	=	Fuel vent system weight scaling factor
k_{28}	=	Fuel control system weight scaling factor
k_{29}	=	Refueling system weight scaling factor
k_{30}	=	Fuel dumping system weight scaling factor
k_{31}	=	Fuel tank foam weight scaling factor
k_{32}	=	Fuel tank seals and sealant weight scaling factor
k_{33}	=	Fuel cell weight scaling factor
k_{34}	=	Hydraulic and pneumatic system weight scaling factor
k_{35}	=	Electrical system weight scaling factor
k_{36}	=	Furnishings weight scaling factor
k_{37}	=	Air conditioning and anti-icing system weight scaling factor – function of wing span
k_{38}	=	Air conditioning and anti-icing system weight scaling factor – function of number of passengers
k_{39}	=	Auxiliary gear weight scaling factor
L	=	Wing structural semi-span
L_{Body}	=	Body length
LEMAC	=	Wing leading edge MAC station location
L_{engine}	=	Engine length
L_{field}	=	Landing field length

Table 2-1. Definition of Terms – Vehicle Synthesis, Contd

L_H	=	Horizontal tail arm
$L_{main\ oleo}$	=	Main landing gear oleo length
$L_{nacelle}$	=	Engine nacelle length
$L_{nose\ oleo}$	=	Nose landing gear oleo length
L_{SP}	=	Spoiler length
L_V	=	Vertical tail arm
n	=	Ultimate flight design load factor
N_{land}	=	Landing load factor
N_T	=	Number of fuel tanks
$N_{Z(gust)}$	=	Ultimate gust load factor
P_{CH}	=	Chord loading
Q	=	Fuel quantity
q_{max}	=	Maximum dynamic pressure
r	=	Ratio of tip thickness to root thickness
R	=	Approximate center of pressure location as fraction of semi-span
RANGE	=	Total range

Table 2-1. Definition of Terms – Vehicle Synthesis, Contd

$S_{Body(wet)}$	=	Wetted area of body
S_f	=	Area of flaps
SFC_{cl}	=	Specific fuel consumption — climb
SFC_{cr}	=	Specific fuel consumption — cruise
S_{fold}	=	Fold wing area
S_H	=	Horizontal tail area
S_{LE}	=	Wing leading edge device area
$S_{main\ oleo}$	=	Main landing gear oleo stroke length
$S_{nose\ oleo}$	=	Nose landing gear oleo stroke length
S_V	=	Vertical tail area
S_w	=	Theoretical wing area
$S_{w(exposed)}$	=	Exposed wing area
$S_{w(gross)}$	=	Gross wing area
S_{wet}	=	Aircraft total wetted area
$S_{wet(nacelle)}$	=	Engine nacelle wetted area
$S_{w(wet)}$	=	Wing wetted area
t_B	=	Wing thickness at the body
t_{BOX}	=	Thickness of the structural box at the root
t_{BRK}	=	Wing thickness at the break
$(t/c)_{BRK}$	=	Thickness to chord ratio at the break

Table 2-1. Definition of Terms — Vehicle Synthesis, Contd

$(t/c)_f$	=	Thickness to chord ratio of the flap
$(t/c)_{\bar{c}w}$	=	Thickness to chord ratio at the MAC
t_{cl}	=	Climb time
$(t/c)_N$	=	Thickness to chord ratio at any spanwise location
T_{cr}	=	Cruise thrust
$(t/c)_r$	=	Thickness to chord ratio at the root
$(t/c)_{SP}$	=	Thickness to chord ratio of the spoiler
$(t/c)_t$	=	Thickness to chord ratio at the tip
$t_{\bar{c}w}$	=	Wing thickness at the MAC
$(t/c)_V$	=	Thickness to chord ratio of the vertical fin
T_{eng}	=	Engine thrust
t_H	=	Thickness of the horizontal stabilizer
t_N	=	Thickness at any spanwise location
t_r	=	Wing thickness at the root
T_R	=	Thrust ratio — design to reference engine
T_{ref}	=	Reference engine thrust
t_t	=	Wing thickness at the tip
$(T/W)_{climb}$	=	Thrust-to-weight ratio — climb
$(T/W)_{Des}$	=	Thrust-to-weight ratio — design
$(T/W)_G$	=	The greater value of thrust-to-weight ratio, takeoff or climb
$(T/W)_{TO}$	=	Thrust-to-weight ratio — takeoff

Table 2-1. Definition of Terms – Vehicle Synthesis, Contd

V_A	=	Maneuver speed at full power setting — equivalent air speed
V_{Body}	=	Volume of body
V_{cl}	=	Climb speed
V_{cr}	=	Cruise speed
$V_{cr_{max}}$	=	Maximum cruise speed
V_{fuel}	=	Volume of fuel
V_H	=	Maximum speed at full power setting – equivalent air speed
VOL	=	Volume of total exposed wing
VOL_{BOX}	=	Volume of structural box between any two spanwise locations
V_{sink}	=	Sink speed
V_{So}	=	Power-off stall speed
V_{struct}	=	Volume of internal structure
V_{tail}	=	Volume of the tail
V_{wing}	=	Volume of the wing
$W_{a/a}$	=	Weight of air conditioning and anti-icing system
W_{atf}	=	Weight of landing gear axles, trunnions, and fittings
W_{aux}	=	Weight of auxiliary gear
W_{bac}	=	Summation of weights of all aircraft items classified under body and contents
W_{Body}	=	Weight of body
W_{brakes}	=	Weight of brakes

Table 2-1. Definition of Terms – Vehicle Synthesis, Contd

$W_{brake\ mech}$	=	Weight of brake mechanisms
W_{cells}	=	Weight of fuel cells
W_{contr}	=	Weight of fuel controls
W_{cruise}	=	Cruise weight
W_{Des}	=	Design mission weight
W_{Dist}	=	Weight of fuel distribution system
W_{drag}	=	Weight of landing gear drag braces
W_{Dump}	=	Weight of fuel dumping system
W_{ec}	=	Weight of engine controls
W_{Elec}	=	Weight of electrical system
W_{eng}	=	Weight of engine
W_{flaps}	=	Weight of flaps
W_{foam}	=	Weight of foam (anti-explosion material)
W_{fold}	=	Weight of fold penalty
W_{fuel}	=	Weight of total fuel
$W_{fuel_{cl}}$	=	Weight of climb fuel
$W_{fuel_{cr}}$	=	Weight of cruise fuel
$W_{fuel_{rs}}$	=	Weight of reserve fuel
W_{furn}	=	Weight of furnishings
W_H	=	Weight of horizontal stabilizer
W_{Hyd}	=	Weight of hydrualic and pneumatic system

Table 2-1. Definition of Terms – Vehicle Synthesis, Contd

W_{land}	=	Landing design weight
W_{LE}	=	Weight of leading edge devices
W_{Lg}	=	Total landing gear weight — alternate method
W_{max}	=	Maximum design weight
W_{oleos}	=	Weight of landing gear oleos
W_{PS}	=	Weight of inlets, cooling, lubrication, and starting system
W_{R}	=	Engine weight ratio - design to reference engine
$W_{Ref\ eng}$	=	Reference engine weight
W_{Refuel}	=	Weight of refueling system
W_{SC}	=	Weight of surface controls
W_{SP}	=	Weight of spoilers
W_{seals}	=	Weight of fuel tank seals and sealant
$(W/S)_{Des}$	=	Design wing loading
$(W/S)_{land}$	=	Wing loading at landing
$(W/S)_{TO}$	=	Wing loading at takeoff
W_{struct}	=	Weight of internal structure
W_{tires}	=	Weight of tires
W_{TO}	=	Takeoff weight
W_{V}	=	Weight of vertical fin
W_{vent}	=	Weight of fuel vent system
W_{wac}	=	Summation of weights of all aircraft items classified under wing and contents
W_{wheels}	=	Weight of wheels

Table 2-1. Definition of Terms — Vehicle Synthesis, Contd

W_{wing}	=	Weight of wing - alternate method
W_{zero}	=	Zero fuel weight
Y	=	Spanwise location of wing relieving loads
y_B	=	Spanwise location of side of body
y_{BRK}	=	Spanwise location of the wing break
$y_{\bar{c}w}$	=	Spanwise location of the MAC
Y_{eng}	=	Spanwise location of engine
y_N	=	Any spanwise location
σ	=	Air density ratio – field
σ_{cl}	=	Average air density ratio - climb
σ_{cr}	=	Air density ratio - cruise altitude
ρ_o	=	Sea level standard air density
ρ_{fuel}	=	Density of fuel
ρ_{struct}	=	Density of structure
λ	=	Wing taper ratio
η	=	Percent semi-span of selected spanwise location
ΣP	=	Summation of wing relieving loads
Δy	=	Spanwise distance between two specific spanwise locations
Δ_{NZ}	=	Incremental gust load factor
$COS\Lambda$	=	Cosine of the wing leading edge sweep angle
$COS\ \Lambda_H$	=	Cosine of the horizontal stabilizer leading edge sweep angle
$COS\Lambda_V$	=	Cosine of the vertical fin leading edge sweep angle

2.1.1 GEOMETRY ANALYSIS. This section describes the calculation of the aircraft dimensions, areas, volume, and miscellaneous other geometry. The data is used to construct a configuration three-view drawing, calculate aerodynamic drag, calculate preliminary loads, and predict the weight. Provisions have been designed into the program to allow the user the ability to easily modify or expand the program as the need arises.

2.1.1.1 Design Weights. Many significant dimensions are derived from imposed load conditions and are hence a function of vehicle design weights. Although the design weights are subsequently modified by program iterations through the weight subroutines, it was found convenient to locate the design weight equations in the geometry subroutine. Design weights are the vehicle total weights defined by specification and operating requirements that are subsequently used in performance, weight prediction, and loads analyses. Design weights are usually required for combat (design mission), landing, and maximum load. In the program design weights are derived by modifying the mission takeoff weight and applying safety margin factors as required by specifications. The equations are:

Design Landing Weight

$$W_{land} = k_1 [W_{TO} - \text{mission payload} + \text{landing payload} - k_2 (\text{mission fuel}) + \text{landing fuel}]$$

Design Mission Weight

$$W_{Des} = W_{TO} - \text{mission payload} + \text{design payload} - k_3 (\text{mission fuel}) + \text{design fuel}$$

Maximum Design Weight

$$W_{max} = k_4 (W_{TO} - \text{mission payload} + \text{maximum payload} - \text{mission fuel} + \text{maximum fuel})$$

Any of the weights calculated by these equations may be input as fixed values if desired.

2.1.1.2 Wing Geometry. The wing geometry calculations have as their basis a generalized wing configuration description expressed in terms of dimensionless ratios and either of the key sizing parameters, wing loading (W/S) or wing area (S_W). The dimensionless ratios include the aspect ratio (AR), taper ratio (λ), thickness-to-chord ratio (t/c), spar location as a fraction of wing chord, etc.

The following equations are for general geometric data; BT indicates that the calculation applies to the basic trapezoid.

Area

BT $$S_w = \frac{W_{Des}}{(W/S)_{Des}}, \text{ or } S_w = \text{input value}$$

Span

BT $$b = \sqrt{AR\ S_w}$$

Mean Aerodynamic Chord (MAC)

BT $$\bar{C}_w = 4/3 \sqrt{\frac{S_w}{AR}} \left[1 - \frac{\lambda}{(1+\lambda)^2}\right], \text{ or } 2/3 \left[C_r + C_t - \frac{C_r\, C_t}{C_r + C_t}\right]$$

Spanwise Location of MAC

BT $$y_{\bar{C}w} = \frac{b}{6} \left[\frac{1 + 2\lambda}{1 + \lambda}\right], \text{ or } \frac{b}{6} \left[\frac{C_r + 2C_t}{C_r + C_t}\right]$$

Chord at Root

BT $$C_r = \frac{2}{1 + \lambda} \sqrt{\frac{S_w}{AR}}, \text{ or } \frac{2\, S_w}{b\,(1 + \lambda)}$$

Thickness at Root

BT $$t_r = (t/c)_r\ C_r$$

Chord at Tip

BT $$C_t = C_r\ \lambda$$

Thickness at Tip

BT $$t_t = (t/c)_t\ C_t$$

Chord at Planform or Profile Break

$$C_{BRK} = C_r \left[1 - \frac{Y_{BRK}}{b/2} (1 - \lambda) \right]$$

Thickness at Break

$$t_{BRK} = (t/c)_{BRK} \; C_{BRK}$$

Thickness at MAC

$$t_{\bar{C}w} = t_{BRK} - \frac{y_{\bar{C}w} - y_{BRK}}{(b/2) - y_{BRK}} (t_{BRK} - t_t), \text{ outboard of break, or}$$

$$t_{\bar{C}w} = t_r - \frac{y_{\bar{C}w}}{y_{BRK}} (t_r - t_{BRK}), \text{ inboard of break}$$

Thickness/Chord Ratio at MAC

$$(t/c)_{\bar{C}w} = \frac{t_{\bar{C}w}}{\bar{C}w}$$

Gross Wing Area

$$S_{w(gross)} = S_w + y_{BRK} (C_{r(extended)} - C_r)$$

Any Spanwise Location

$$y_N = \eta \, (b/2), \text{ or } y_N = \text{input value}$$

Chord at Any Spanwise Location

$$C_N = C_r \left[1 - \frac{y_N}{(b/2)} (1 - \lambda) \right] \quad \text{, outboard of break, or}$$

$$C_N = C_r \left[1 - \frac{y_N}{(b/2)} (1 - \lambda) \right] + (C_{r\ (\text{extended})} - C_r) \left[\frac{y_{BRK} - y_N}{y_{BRK}} \right] ,$$

inboard of break

Thickness at Any Spanwise Location

$$t_N = t_{BRK} - \frac{y_N - y_{BRK}}{(b/2) - y_{BRK}} (t_{BRK} - t_t), \text{ outboard of break, or}$$

$$t_N = t_r - \frac{y_N}{y_{BRK}} (t_r - t_{BRK}), \text{ inboard of break}$$

Thickness/Chord Ratio at Any Spanwise Location

$$(t/c)_N = \frac{t_N}{C_N}$$

Wing Volumes and Cross-Sectional Areas

a. Cross-sectional area of structural box at any spanwise location,

$$A_{c.s.Box} = 0.91 \left[(\text{decimal R.S. loc}) - (\text{decimal F.S. loc}) \right] C_N t_N$$

b. Volume of structural box between any two spanwise locations

$$Vol_{Box} = \frac{\Delta y}{3} \left[A_{c.s.Box_1} + \sqrt{A_{c.s.Box_1} A_{c.s.Box_2}} + A_{c.s.Box_2} \right]$$

c. Cross-sectional area of total wing at any spanwise location

$$A_{c.s.} = 0.68\, C_N\, t_N$$

d. Volume of total exposed wing (outside body width, B)

$$\text{Vol} = \frac{0.68}{3} \{(b-B) [C_B\, t_B + (C_B + C_t)\,(t_B + t_t) + C_t\, t_t]\}$$

2.1.1.3 Tail Geometry. Tail geometry calculations are made in a manner similar to those of the wing. The major exception is the derivation of the tail areas based on tail volume coefficients and the previously computed tail arm.

Areas

$$S_H = C_{H_{Vol}} \quad S_w \left[\frac{\bar{C}_w}{L_H} \right]$$

$$S_V = C_{V_{Vol}} \quad S_w \left[\frac{b}{L_V} \right]$$

2.1.1.4 Fuselage Geometry. The dimensional data for the fuselage presents a problem because of the variety of possible configurations. For supersonic fighter aircraft, a significant step toward the solution of this problem has been made in the recent work of Caddell (Reference 4). Following are some of the equations adapted from this study. For the case of a passenger transport, the fuselage may be assumed to be cylindrical for much of its length with a cross section established by the passenger seating arrangement. Thus, the length, volume, and wetted area become a function of the number of seat rows.

{ Volume less fuel and structural volume equals weight less fuel and structure weight divided by density of remaining items, (empirically derived) }

$$V_{fuel} = \frac{W_{fuel}}{\rho_{fuel}}$$

$$V_{struct} = \frac{W_{struct}}{\rho_{struct}}$$

Length

$$L_{Body} = k_5 \left[\frac{\left(F.R._{B(equiv.)}\right)^2}{\pi/4} \quad \left(\frac{\Sigma V}{0.7}\right) \right]^{1/3}$$

where

F.R. = fineness ratio, or

L_{Body} may be an input value and the equivalent fineness ratio can then be computed by solving the L_{Body} equation.

Maximum Cross-Sectional Area of Total Airplane

$$A_{max} = \frac{\Sigma \text{ Vol}}{0.7 \, L_{Body}}$$

Volume of Body

$$V_{Body} = \Sigma \text{ Vol} - V_{wing} - V_{tail}$$

Equivalent Diameter of Body

$$D_{Body} = \sqrt{\frac{V_{Body}}{0.7 \, (\pi/4) \, L_{Body}}}$$

Body Fineness Ratio

$$F.R._{B(equiv)} = \frac{L_{Body}}{D_{Body}}$$

Maximum Cross-sectional Area of Body

$$A_{Body(max)} = (\pi/4) \, (D_{Body})^2$$

Wetted Area of Body

$$S_{Body(wet)} = k_6(3.309)\,(L_{Body}\,V_{Body})^{1/2} - 2\,C_B t_B$$

2.1.1.5 Landing Gear. The dimensions required for computation of landing gear geometry are the length and stroke, and wheel and tire sizes. Unless given, these dimensions are derived from aircraft configuration requirements and landing gear loads. Therefore, calculations of approximate landing gear loads are included in the geometry calculations.

The conditions that establish the oleo length include the location of the mounting point, aircraft ground clearance, and the length requirements to accommodate the oleo stroke. The stroke is established by the landing energy absorption requirements and a landing load factor. The landing load factor and landing weight are used to compute the loads, and the landing stall speed or an approximation is used to establish the kinetic energy to be dissipated through braking.

Main Oleo Length

$$L_{main\ oleo} = k_7\,L_{body}$$

Main Oleo Stroke

$$S_{main\ oleo} = \frac{12}{g}\left[\frac{\frac{1}{2}(V_{sink})^2}{(N_{land}-1)}\right]$$

Vertical Load on Main Landing Gear

$$F_{V\ main} = \frac{W_{land}}{2}\,(N_{land}-1)$$

Drag Load on Main Landing Gear

$$F_{D\ main} = 0.481\,F_{V\ main}$$

Vertical Load on Nose Landing Gear

$$F_{V\ nose} = 0.4\ F_{V\ main}$$

Drag Load on Nose Landing Gear

$$F_{D\ nose} = 0.481\ F_{V\ nose}$$

Kinetic Energy per Brake

$$\text{K.E. per Brake} = \frac{\frac{1}{2} W_{Des} V_{so}^{2}}{g\,(\text{No. Wheels})}$$

$$\text{Main Wheel Diameter} = 1.224 \sqrt{\frac{W_{land}\ N_{land}}{500\,(\text{No. Wheels})}}$$

$$\text{Wheel Flange Width} = 0.863\,(\text{Diameter})$$

$$\text{Tire Diameter} = \frac{\text{Wheel Diameter}}{1.4} + 2\,(\text{Wheel Flange Width})$$

$$\text{Nose Wheel Diameter} = 1.224 \sqrt{\frac{W_{land}\ N_{land}}{5000\,(\text{No. Wheels})}}$$

Nose Oleo Length

$$L_{nose\ oleo} = 0.9\ L_{main\ oleo}$$

Nose Oleo Stroke

$$S_{nose\ oleo} = \frac{L_{nose\ oleo}}{4.5}$$

2.1.1.6 Nacelle Geometry. On aircraft with externally mounted engines, the nacelle dimensions and areas depend entirely on individual mounting and cowling arrangements, and a generalization is difficult. The following are some elementary equations that may be used.

Diameter

$$D_{nacelle} = k_8 D_{engine}$$

Length

$$L_{nacelle} = k_9 L_{engine}$$

Maximum Cross-Sectional Area

$$A_{c.s.(nacelle)} = \pi/4 \, D^2_{nacelle}$$

Wetted Area

$$S_{wet(nacelle)} = k_{10} \pi \, D_{nacelle} \, L_{nacelle}$$

2.1.1.7 General Geometry. Computations for the wing exposed area, wing wetted area, and total aircraft wetted area are:

Exposed Wing Area

$$S_{w(exposed)} = (C_B + C_{BRK})(y_{BRK} - y_B) + (C_{BRK} + C_t)\left(\frac{b}{2} - y_{BRK}\right)$$

Wing Wetted Area

$$S_{w(wet)} = \left[2 + (t/c)_{\bar{C}w}^{1.5}\right] S_{w(exposed)}$$

Aircraft Total Wetted Area

$$S_{wet} = \Sigma \, S_{wet\ components}$$

2.1.1.8 Loads Calculations. The following equations reflect a simplified treatment of the vehicle loads including gust load factors. This data is utilized in the weight computation portion of the program.

Wing Normal Force Curve Slope

$$C_{N\alpha_W} = \left[\frac{6}{1 + \dfrac{6}{\pi\, AR}}\right] + 2.15 \left(\frac{S_H}{S_W}\right)$$

Aircraft Inertia Factor — Gust Alleviation

$$U = \frac{2\, W_{Des}}{0.0765\, C_{N\alpha_W}\, \bar{C}_W S_W}$$

Gust Response Factor

$$K = 0.88 \left(\frac{U}{U + 5.3}\right)$$

Maximum Speed

$$V_H = 29 \sqrt{q_{max}}$$

Incremental Gust Load Factor

$$\Delta N_Z = \frac{0.1189\, V_H\, C_{N\alpha_W}\, S_W\, K}{2\, W_{Des}}$$

Ultimate Gust Load Factor

$$N_{Z(gust)} = 1.5\,(1 + \Delta N_Z)$$

Maneuver Speed

$$V_A = 29 \sqrt{\frac{N_{Z(gust)}}{1.5} \frac{W_{Des}}{S_W} \frac{1}{0.25\, C_{N\alpha_W}}}$$

Vertical Tail Normal Force Curve Slope

$$C_{N\beta} = 5.7 \frac{1.5\, AR_V}{1.5\, AR_V + 2}$$

Gust Load on Vertical Tail

$$F_{V(Gust)} = \frac{C_{N\beta}\, 50\, V_H\, S_V}{841}$$

Vertical Tail Normal Force Coefficient

$$C_{N(maneuver)} = 0.22\, C_{N\beta}$$

Maneuver Load on Vertical Tail

$$F_{V(maneuver)} = \frac{C_{N(maneuver)}\, V_A^2\, S_V}{841}$$

Ultimate Vertical Tail Load

$$F_{V(ult)} = 1.5 \quad F_{V(gust)}, \text{ or } 1.5 \quad F_{V(maneuver)}$$

Horizontal Tail Normal Force Curve Slope

$$C_{N\alpha_h} = 5.7 \left[\frac{AR_H}{AR_H + 2}\right]$$

Horizontal Tail Normal Force Coefficient

$$C_{N_h} = 0.147\ C_{N\alpha_h}$$

Maneuver Load on Horizontal Tail

$$F_{H(maneuver)} = \frac{C_{N_h}\ V_A^{\ 2} S_H}{841}$$

Gust Load on Horizontal Tail

$$F_{H(gust)} = \frac{C_{N\alpha_h}\ 50\ V_A S_H}{841}$$

Ultimate Horizontal Tail Load

$$F_{H(ult)} = 1.5\ \ F_{H(gust)},\ \text{or}\ 1.5\ \ F_{H(maneuver)}$$

2.1.2 AREA DISTRIBUTION. Area distribution of the (assumed cylindrical) body geometry that is output from the vehicle synthesis process may be achieved by use of the area distribution routines. The reason for deriving a so-called area distributed body shape is to minimize the zero-lift drag buildup through the transonic flight regime. For the purposes of the present analysis a flight Mach number of 1.0 is assumed.

A simplified functional flow diagram of the area distribution process is presented in Figure 2-1. Output from the vehicle synthesis routines comprises a vehicle group weight statement and basic vehicle geometry sufficient to make a three-view general arrangement drawing of the vehicle. Included in the geometry output are wetted areas and volumes of major components. This data is input to the aircraft balance routine, which locates the centers of gravity for each of the elements making up the group weight statement. If the user selects an area distributed fuselage shape, the outputs from both the vehicle synthesis and balance routines are used to drive the area distribution process.

The primary steps involved in generating an area distributed body shape are as follows. Cross-sectional areas of major components are computed at specified longitudinal

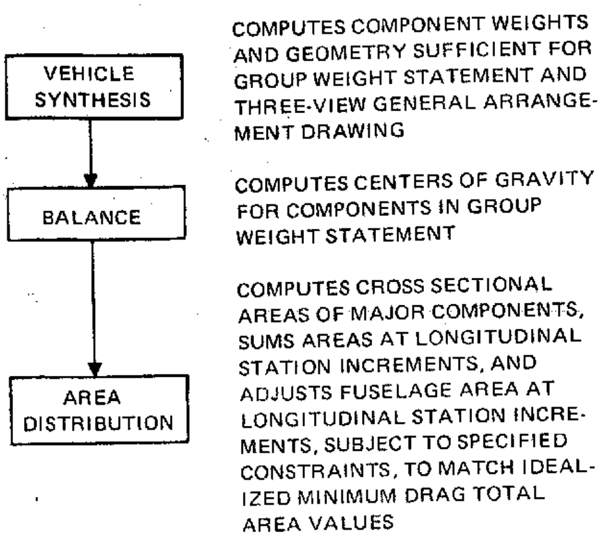


Figure 2-1. Functional Flow Diagram of Area Distribution Process

fuselage station increments. At each station a total area is derived by summing the areas of the individual components. This total area is compared to values for total area taken from an idealized, minimum drag plot of area versus station. An attempt is made to match the idealized value for total area by adjusting the area of the fuselage at each station (subject to specified minimum and maximum body area constraints), while holding the areas of the remaining components constant.

An example of a plot showing cross sectional area versus fuselage station for a typical fighter type aircraft is shown in Figure 2-2. The current version of the program computes area values for the surfaces (wing, horizontal, and vertical) and the fuselage. The remaining components (inlets, canopy, engine pods and pylons, and miscellaneous fairings) were left for future work. Figure 2-3 illustrates the actual program logic and data flow.

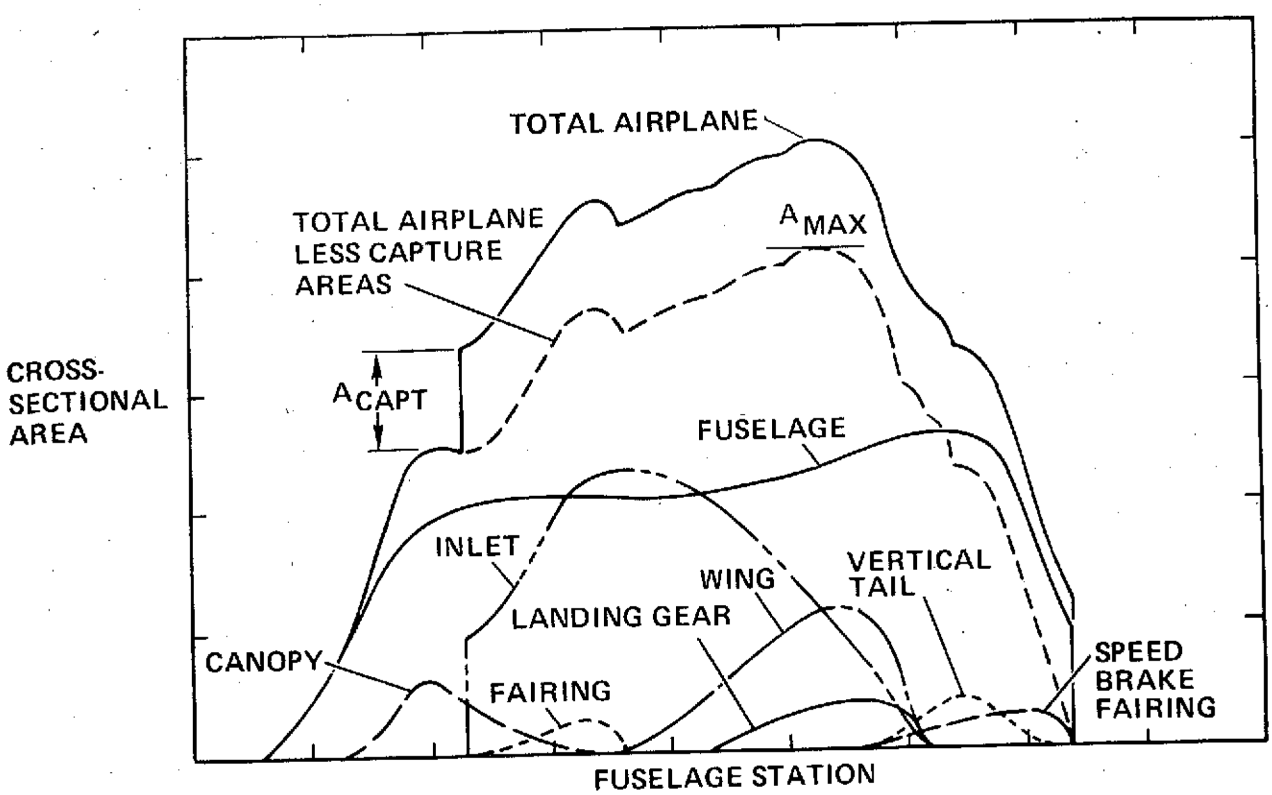


Figure 2-2. Typical Example of a Cross-Sectional Area Distribution Plot

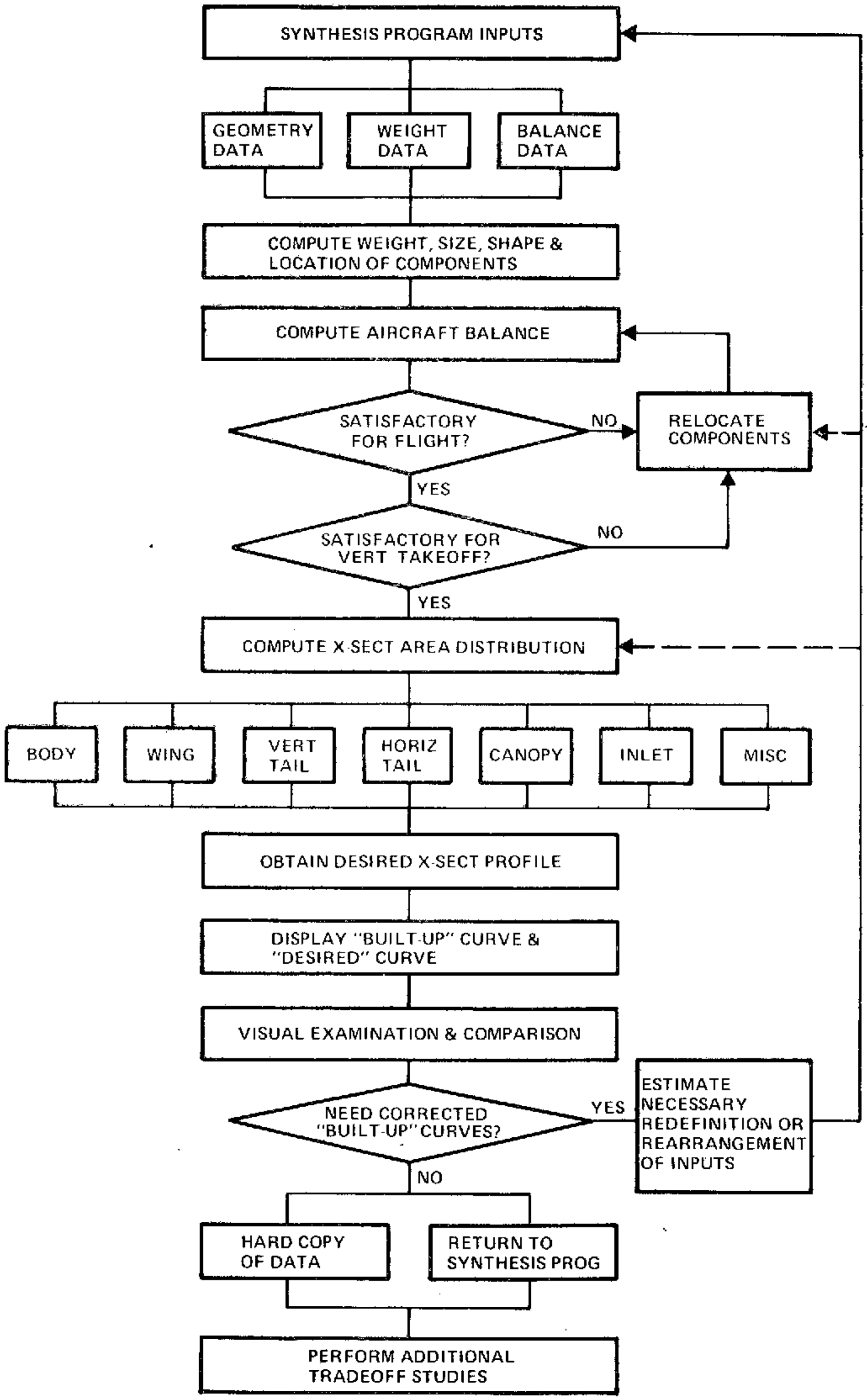


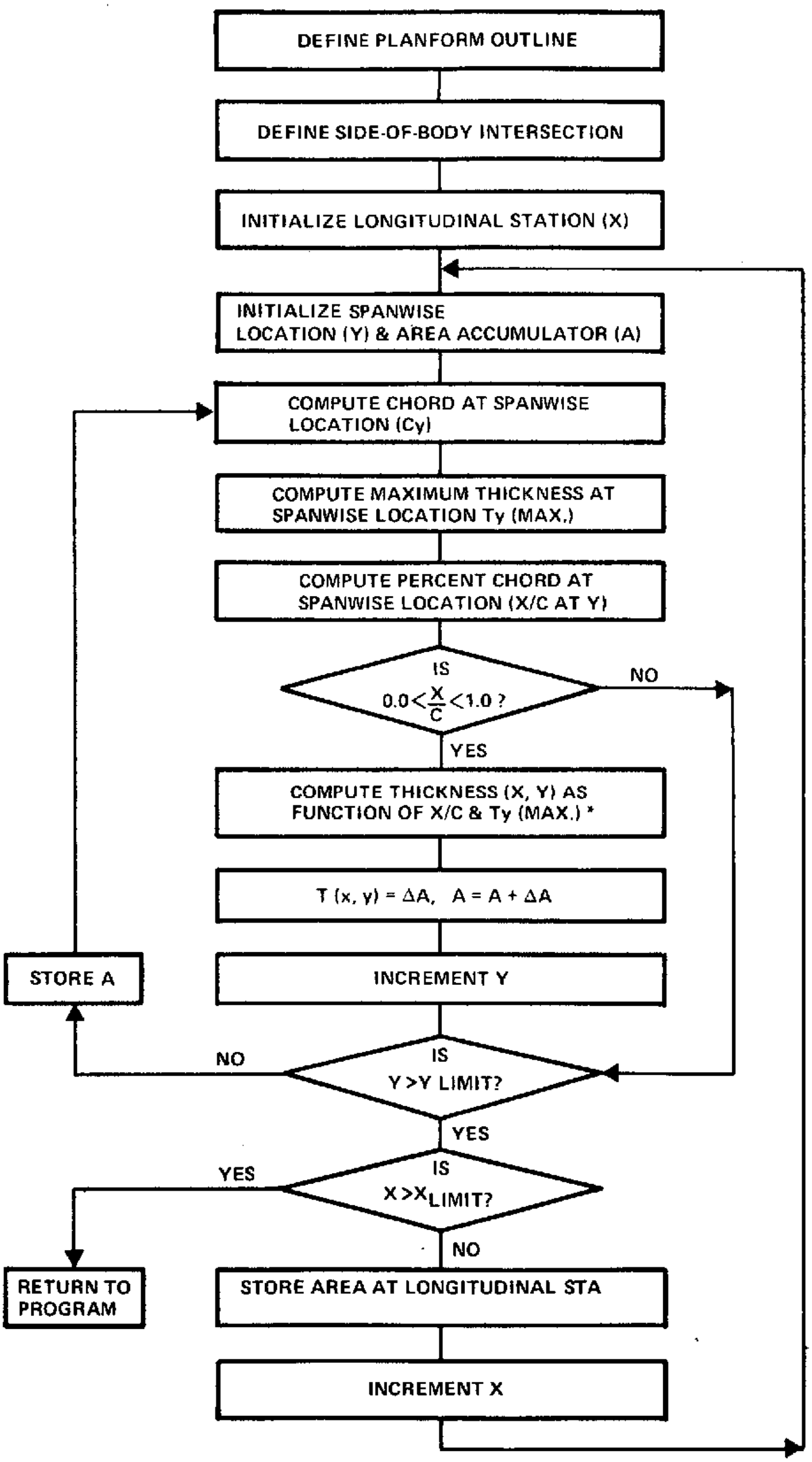
Figure 2-3. Area Distribution Logic and Data Flow

Data necessary to compute the cross-sectional areas of airfoil surfaces includes basic planform geometry (area, span, sweep, taper ratio, root chord, etc.), thickness ratios at the root and tip, longitudinal location of wing leading edge and fuselage centerline intersection, and spanwise location of the wing and side of body intersection. At each longitudinal fuselage station, the airfoil thicknesses at one-inch spanwise increments are computed. Thicknesses at each station are then integrated in a spanwise direction between the inboard and outboard boundaries of the surface to determine cross-sectional areas. Figure 2-4 summarizes the flow through the airfoil area computation process. Figure 2-5 illustrates a computed area distribution for typical airfoils and shows the effects of sweep and section profile.

The initial shape of the fuselage is assumed to be cylindrical. A typical fighter body is represented by a tapered nose section along 30% of the body length, and a tapered tail section, driven by engine diameter, along the remaining 70%. Transport bodies are represented by nose and tail tapered sections each along 30% of the body length. Areas are computed by assuming a circular cross-section at each station. Fuselage length, maximum width (diameter), and fuselage volume are brought across from the vehicle synthesis routines.

A non-dimensionalized plot of actual cross sectional area versus longitudinal fuselage station can be derived as follows. The areas from the individual component distributions are summed at each station. The percent of maximum area is curve fit as a function of percent of fuselage length. The result is a smooth curve similar to the one shown in Figure 2-6A. This initial built-up curve is mathematically compared to a so-called desired curve, Figure 2-6B. The desired curve is an idealized representation of a vehicle of similar performance and mission whose profile exhibits minimum drag characteristics. The curve is derived on the basis of required aircraft length, volume, maximum cross-sectional area, and the longitudinal location of the maximum area section.

A revised built-up curve is generated by adjusting the initial curve subject to specified constraints. Changes to the initial area distribution curve are made by allowing one component of the total area, that of the fuselage, to change while the rest are held constant. Constraints are set on the amount that the fuselage area is allowed to change; consequently, the revised curve, Figure 2-6C, will not necessarily match the ideal curve exactly. Constraints on the fuselage area are made by specifying a maximum and minimum allowable area between given fuselage stations. Any number of these constraints may be selected. By using small increments along the fuselage length it is possible to closely control the allowed area variation, or by merely defining a single maximum and minimum value, an approximate area distributed fuselage can be defined with a minimum of input. Factors that influence the use of these constraints include requirements for a minimum cabin width to allow the use of a given side-by-side seating or cargo pallet arrangement; a minimum fuselage diameter to allow the use of a



* STORED NONDIMENSIONAL EQUATIONS FOR DIFFERENT THICKNESS DISTRIBUTIONS, WHERE T (x,y) IS A FUNCTION OF T/C & Ty (MAX.)

Figure 2-4. Airfoil Area Distribution Analysis Routine

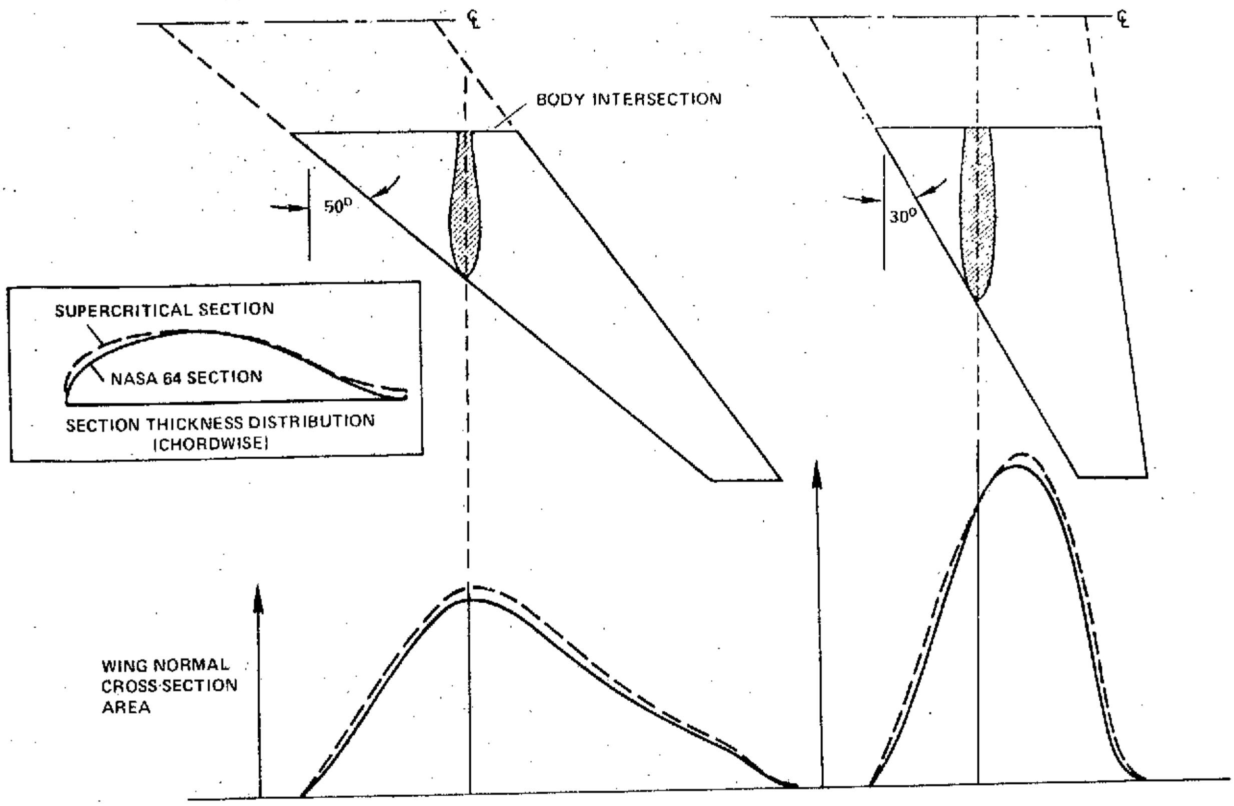
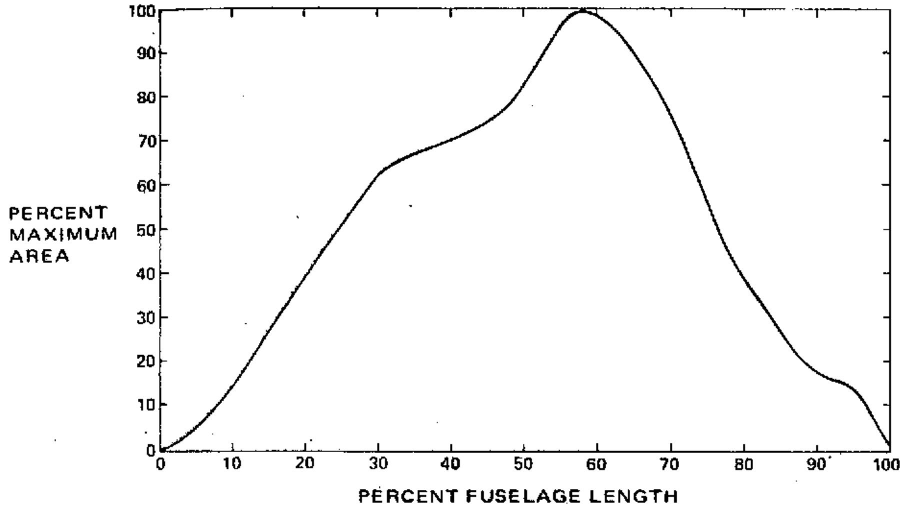


Figure 2-5. Effect of Sweep and Section Profile on Wing Area Distribution

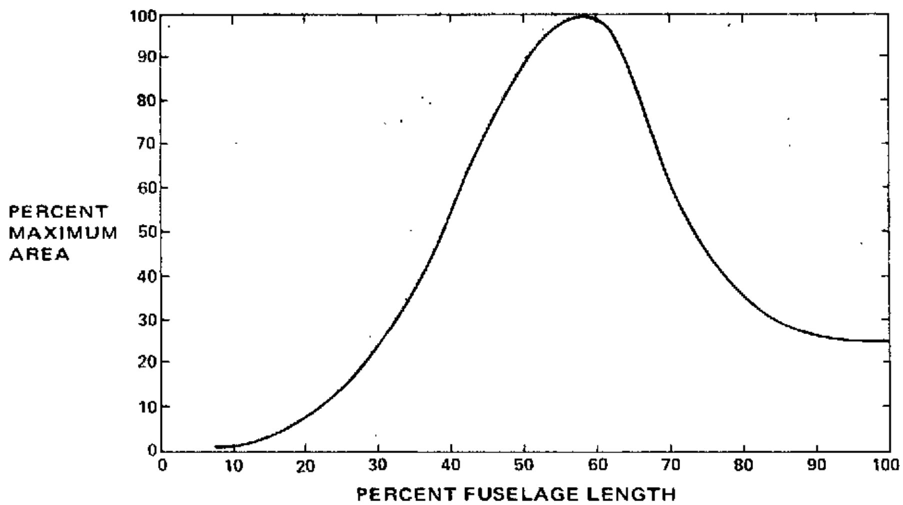
given engine; a maximum fuselage width to allow use of existing carrier elevators or boarding ramps; or various clearance requirements for installation of special mission equipment, aircraft handling, or cargo loading procedures.

If a satisfactory body shape is not achieved for a given vehicle within the specified constraints, the user has any or all of the following alternatives. The user may change the constraints to allow a new range of fuselage area variation, or the user may return to the balance portion of the program and relocate individual components in a logitudinal direction, or the user may return to the vehicle synthesis portion of the program to make changes affecting the overall vehicle geometry and weight. By use of a trial and error process it is possible for the user to establish an optimum vehicle body shape for a given mission.

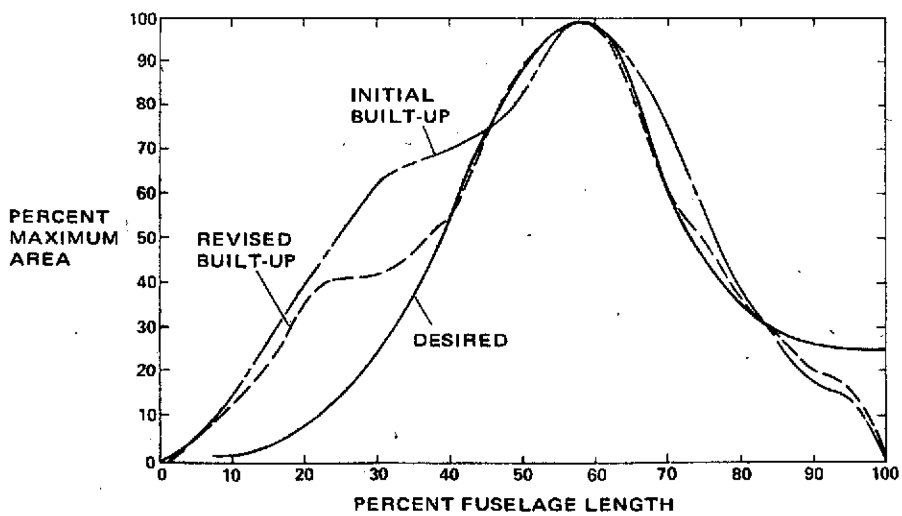
The most efficient use of the area distribution process can be made utilizing the interactive graphics mode of operation. Displays have been programmed to allow the user to view both the initial and revised area distribution plots overlayed on the idealized curve, and a plan view of the resultant area ruled aircraft. Changes that can be implemented from the console include changes to the fuselage area variation constraints, changes to the balance routine input, and changes to the vehicle synthesis inputs with a corresponding resizing of the vehicle. In this way a systematic approach to deriving the best possible fuselage shape may be accomplished, with the ability of the user to immediately examine the effects of each change.



A. INITIAL BUILT-UP AREA PROFILE CURVE



B. DESIRED AREA PROFILE CURVE



C. REVISED BUILT-UP AREA PROFILE CURVE

Figure 2-6. Computerized Area Profile Curves

2.1.3 WEIGHT ANALYSIS. This section describes the weight calculations that are needed to produce a group weight statement and the gross weights required for the aircraft performance analyses. The equations listed here represent a complete and logical approach to aircraft weight determination; however, they are not the only ones that could apply. But they are representative, and may be modified or replaced if need be.

2.1.3.1 Wing Weight. The wing weight is estimated in terms of the basic wing box plus penalties, which are combined to obtain total wing weight. The basic wing box is comprised of ribs, and fixed leading and trailing edges. Penalty items include the high-lift devices, spoiler, and wing fold. The equations are based on those of Reference 5. The original computation required the selection of a value for the constant C_2 that was based on an allowable compressive stress at wing root for the type of structure being considered. Calculations have been added to provide a value for the constant C_2 based on a cursory bending analysis of the wing.

Approximation of Root Bending Moment

$$BM = \frac{1}{2} W_{DES} \; n \; \left[0.43\,(b/2) - \frac{1}{2}\,(\text{body width}) \right]$$

Chord Loading

$$P_{CH} = \frac{BM}{0.85\,(t_{box})\,(C_{box})}$$

Allowable Bending Stress at Wing Root (See Figure 2-1 in Reference 3)

$$f_a = k_{11} \left(\frac{P_{CH}}{P_{CH} + 3000} \right)$$

Equation Constant (See Figure 2-2 in Reference 3)

$$C_2 = \frac{2.215}{(f_a)^{0.935}}$$

Term Accounting for Wing Relieving Loads (See Figure 2-3 in Reference 3)

$$\Sigma\,(P_{CH} Y^2 k_{12}) \quad \text{where } k_{12} = f(t_p/t_R)$$

Basic Wing Equation

$$W_{wing} = \frac{(3 + r)\left[C_1 S_w (nW)^{0.25} + \frac{C_2 n}{t_r}\left(\frac{WRL^2}{1/R + r} - \Sigma(PY^2 k_{12})\right)\right]}{(3 + r) + \frac{C_2 n}{t_r}(0.34L^2)}$$

where

$L = \frac{(b/2)}{\cos \Lambda}$, structural semispan.

$R = \frac{1}{2(1+\lambda)} - \frac{(1-\lambda)}{3(1+\lambda)} + \frac{2}{3\pi}$, approximation of location of the center of pressure as a ratio of semispan (see Figure 2-4 in Reference 3).

$r = (t_t/t_r)$, ratio of tip thickness to root thickness.

Additional weight penalties for special features should be assessed based on an individual investigation of each feature. If no special features are defined but some may be expected, then a general allowance should be made.

Flap Weight Equation (See Figure 2-5 in Reference 3)

$$W_{flaps} = k_{13}\left[\frac{W_{Des}\, S_f\, C_f}{100,000\, S_w}\right]^{0.57}\left[\frac{1}{(t/c)_f}\right]^{0.228}\left[0.5 + 0.3\left(\frac{\text{fowler}}{0.5}\right) + 0.2\left(\frac{\text{deflection}}{37}\right)\right]$$

The term "$(t/c)_f$" refers to the thickness ratio of the wing at the flap midspan, "fowler" refers to fowler motion as a ratio to flap chord, and "deflection" is flap deflection in degrees. The last element, [0.5 + 0.3 (fowler/0.5) + 0.2 (deflection/37)], has been added to the equation to provide some sensitivity of the weight to flowler motion and deflection. The term is based on several informal studies indicating that, based on a flap with 50% fowler motion and a 37-degree deflection, about 30% of the weight penalty would vary with fowler motion and about 20% would vary with deflection. The differences in flap configuration are accounted for by selection of the constant (k_{13}).

Leading Edge Device Weight (See Figure 2-6 in Reference 3)

$$W_{LE} = k_{14} \left[\frac{W_{Des}\, n}{1000\, S_w} \right]^{0.324} (S_{LE})^{1.081}$$

Spoiler Weight (See Figure 2-7 in Reference 3)

$$W_{SP} = k_{15} \left[\frac{W_{Des}\, L_{SP}}{1000\, S_w} \right]^{0.569} (C_{SP})^{1.139} \left[\frac{1}{(t/c)_{SP}} \right]^{0.228}$$

Fold Penalty Weight (See Figure 2-8 in Reference 3)

$$W_{FOLD} = k_{16} \left(\frac{S_{fold}}{S_w} \right)^{1/2} W_{Wing}$$

The wing fold weight penalty was derived using total wing weight as a parameter. In the program the equations are arranged to use the wing weight prior to the addition of the fold weight penalty.

Total Wing Weight Alternative Method

$$W_{Wing} = k_{17} \left[\frac{W_{Des}\, n}{1000} \right]^{0.52} (S_w)^{0.7} (AR)^{0.47} \left[\frac{(1+\lambda)}{(t/c)_{\bar{c}_w}} \right]^{0.4} \left[0.5 + 0.5 \left(\frac{1}{\cos \Lambda} \right) + 0.905 \left(\frac{1}{\cos \Lambda} - 1 \right)^2 \right]$$

This equation offers an alternative method of computing wing weight, which, though less sensitive to some configuration details, may be desirable during early aircraft concept development.

2.1.3.2 Tail Weight. Tail weight equations are:

Horizontal Tail (See Figure 2-9 in Reference 3)

$$W_H = k_{18} \left[\frac{W_{Des} \bar{C}_w b_H S_H}{L_H t_H \operatorname{Cos} \Lambda_H} \times 10^{-9} \right]^{1/2}$$

Vertical Tail (See Figure 2-10 in Reference 3)

$$W_V = k_{19} \left[\left(\frac{W_{Des}}{S_w} \right)^{0.8} \left(\frac{b}{L_V} \right) \frac{S_V \; AR_V}{100 \, (t/c)_V^{1/2} \cos^2 \Lambda_V} \right]^{0.9853}$$

The tail equations do not include balance weights for the rudder or elevator, and do not include any special features such as "T" tail arrangements or folding. The constants, k_{18} and k_{19}, may be raised to account for some features, or new equations could be developed and added to the program as the need arises.

2.1.3.3 Fuselage Weight. The following equation yields good results in many cases. However, the equation does not provide sensitivity to the detailed fuselage arrangement.

$$W_{Body} = k_{20} \, (n \; W_{Des})^{0.3} \, (L_{Body})^{0.9} \, (B + H)^{1.05} \, (q_{max})^{0.1775}$$

2.1.3.4 Landing Gear Weight. The landing gear weight can be developed in several ways, depending on available data and configuration definition. The following equations provide a selection.

Brakes

$$W_{brakes} = (5.62 \times 10^{-4}) \, (K.E.)^{0.75} \; (\text{Number of Wheels})$$

Wheels

$$W_{wheels} = (0.7) \, (\text{No. wheels}) \left[\frac{W_{land} \, N_{land}}{(\text{No. wheels}) \, 1000} \right]^{0.7}$$

$$(\text{Diam})^{1.3} \left[(1 + 0.025) \, (\text{width}) \right]$$

Tires

$$W_{tires} = \frac{W_{land} N_{land}}{1000}\ \text{, main or}\ \frac{W_{land} N_{land}}{10,000}\ \text{, nose}$$

Brake Mechanism

$$W_{brake\ mech} = (2.2) \left[\frac{W_{brakes}}{\text{No. wheels}} \right]^{0.5} \text{(No. wheels)}$$

Oleos

$$W_{oleos} = (0.0016)\,(F_{V_{main}}\ S_{main\ oleo})^{0.75} \left(\frac{L_{main\ oleos}}{S_{main\ oleo}} \right) \text{(No. oleos)}$$

Axles, Trunnions, and Fittings

$$W_{atf} = (0.0015)\,(F_{V_{main}})\ \text{(No. oleos)}$$

Drag Braces

$$W_{drag} = (0.0013)(F_{D_{main}})\ \text{(No. oleos)}$$

Total Landing Gear Weight - Alternative Method

$$W_{LG} = k_m W_{max} + k_{des} W_{des} + k_{land} W_{land} + \text{(constant)}$$

2.1.3.5 Surface Controls Weight. (See Figure 2-11 in Reference 3)

$$W_{SC} = k_{21} \left(L_{body} + \frac{b}{\cos \Lambda} \right) (W_{Des})^{0.3} (n)^{0.64}$$

In the equation for surface controls, the weight is assumed to be a function of the dimensions of the aircraft, primarily, and, to a lesser degree, of the loads and overall complexity as expressed by the gross weight. The load factor term provides a means of accounting for intangible factors (as maneuverability and system rigidity) that lack a quantitative means of expression.

The following equation provides an alternative means of scaling the surface control weight. This equation is useful when a nominal system weight has been derived for a sample aircraft, and the value needs to be scaled for the synthesized aircraft.

$$W_{SC} = k_{22}\,(S_W + S_H + S_V) + (\text{constant})$$

2.1.3.6 Nacelle Weight. Nacelle weight is either input as a fixed value or may be scaled by the same means as the engines. (See Figure 2-12 in Reference 3.) It is anticipated that additional development work will be carried out in this area in the future, and the resulting equations incorporated into the program.

2.1.3.7 Engine Weight. Engine weight is frequently a fixed input value because aircraft configuration studies are often done with a specific engine in mind. However, provisions have been made, by the following equations, to scale engine weight from a given reference engine weight, and thereby allow for variation about a basic type of engine. When scaling, the thrust required and the thrust ratio can be determined by the following equations.

$$T_{eng} = (T/W)_{TO}\,(W_{TO})$$

$$T_R = \frac{T_{eng}}{T_{ref}}$$

Then, the engine weight ratio is computed from the equation:

$$W_R = k_{e_1}\,(T_R) + k_{e_2} \qquad \text{(See Figure 2-13 in Reference 3.)}$$

With the engine reference weight and reference thrust given, and the weight ratio computed, the scaled engine weight is calculated by the equation:

$$W_{eng} = W_R\,(W_{ref})$$

2.1.3.8 Propulsion Systems Weight. Propulsion systems weight equations are:

Inlets, Cooling, Lubrication, Starting

$$W_{PS} = k_{23}\,(T_{eng})$$

This equation assumes all of the items are a function of thrust. This assumption is valid in most cases but requires caution in the case of inlets. Inlets can be very complex for supersonic aircraft, and the future development of a more sensitive expression would be useful.

Engine Controls

$$W_{EC} = k_{24} (Y_{eng} + 0.3 L_{Body}) \text{ (No. Engines) + constant}$$

where

$$Y_{Eng} = \text{the spanwise engine location}$$

This equation relates engine control weight to the distance from cockpit to engines.

2.1.3.9 Fuel System Weight. The fuel system weight is calculated by a series of equations that were first presented in Reference 6. The parameters used are fuel quantity Q, number of tanks N_T, number of engines N_E, and engine thrust T.

Distribution System

$$W_{Dist} = k_{25} T_{eng}^{0.5} + k_{26} Q^{0.7} N_T^{0.25}$$

Vent System

$$W_{Vent} = k_{27} \frac{N_T T_{eng}}{1000}$$

Fuel Controls

$$W_{Contr} = k_{28} (N_T Q)^{0.5}$$

Refueling System

$$W_{Refuel} = k_{29} N_T^{0.5} Q^{0.33}$$

Dumping System

$$W_{Dump} = k_{30} Q^{0.65}$$

Foam (Antiexplosion Material)

$$W_{Foam} = k_{31} Q$$

Seals and Sealant

$$W_{Seals} = k_{32} \; N_T^{\;0.25} \; Q^{\;0.75}$$

Fuel Cells

$$W_{Cells} = k_{33} \text{ (pounds of fuel in cells)}$$

The total fuel system is the sum of preceeding eight equations.

2.1.3.10 Systems and Equipment Weight. Computation of systems and equipment weight is accomplished by scaling certain base parameters. While the method provides a simple and reasonable means of determining weight, the program incorporates a capability for easily updating and expanding the analysis as more sensitive equations are developed.

Hydraulic and Pneumatic System (See Figure 2-13 in Reference 3)

$$W_{Hyd} = k_{34} \; (W_{Des})^{0.5} \; (L_{Body} + b)$$

Electrical System

$$W_{Elec} = k_{35} \, W_{Des}$$

Furnishings

$$W_{Furn} = k_{36} \text{ (No. Passengers) + fixed weight}$$

Air Conditioning and Anti-Icing

$$W_{a/a} = k_{37}\, b + k_{38}\,(\text{No. Passengers}) + \text{fixed weight}$$

Auxiliary Gear

$$W_{Aux} = k_{39}\, W_{Des}$$

2.1.4 PERFORMANCE ANALYSIS. This section describes the calculations necessary to approximate the performance, propulsion, and loads data required to close the iteration cycle within the aircraft synthesis program. The subroutine is designed to provide wing loadings, thrust-to-weight ratios, fuel weights, and gust load factor. The takeoff weight is updated in the weight subroutine during the iteration cycle. The equations are segregated into five groups, including the data for takeoff and landing condition, climb condition, cruise condition, design condition, and a gust load approximation.

Data developed for the takeoff and landing condition includes lift and drag coefficients, wing loadings, and thrust-to-weight ratio. The wing loading is derived from the landing field length requirement, lift and drag coefficients, and assumed deceleration rates. The lift coefficient is computed as a simple function of aspect ratio. The drag coefficient is computed from an equivalent skin friction drag coefficient and aircraft wetted area. The thrust-to-weight ratio is derived from takeoff thrust requirements for the landing field length specified. The equations are:

Maximum Lift Coefficient

$$C_{L_{max}} = \frac{k_{C_{L_{max}}}}{1 + \frac{3}{AR}}$$

The lift coefficient scaling factor is obtained from $k_{C_{L_{max}}} = C_{L_{max}}\left[1 + \frac{3}{AR}\right]$ for the baseline wing condition. Then as the baseline wing aspect ratio is varied the effect on $C_{L_{max}}$ will be accounted for in the resizing operation.

Coefficient of Lift – Landing

$$C_{L_{land}} = 0.75\,(C_{L_{max}})$$

Coefficient of Drag – Equivalent Profile

$$C_{D_o} = C_{D_f} \left[\frac{S_{wet}}{S_w}\right] \quad (1.1)$$

Wing Loading – Landing

$$(W/S)_{land} = \frac{L_{land}(\sigma)(a)}{420}\left[\frac{L_{field}}{1.67} - GOH_{min}\left(\frac{C_{L_{land}}}{C_{D_o} + \frac{k_{GE}(C_{L_{land}})^2}{\pi(AR)(e)} + \frac{S_w}{10^4}}\right)\right]$$

Wing Loading – Takeoff

$$(W/S)_{TO} = (W/S)_{land}\left[\frac{W_{TO}}{W_{land}}\right]$$

Coefficient of Lift – Takeoff

$$C_{L_{TO}} = 0.65\ (C_{L_{max}})$$

Thrust-to-Weight Ratio – Takeoff

$$(T/W)_{TO} = \frac{1.8}{L_{field}}\left[\frac{420\ (W/S)_{TO}}{C_{L_{TO}}(\sigma)(g)} + 35\right]$$

Data developed for climb conditions include lift-to-drag ratio, thrust-to-weight ratio, speed, distance, time, and fuel weights. The thrust-to-weight ratio is computed for a sustained angle of climb requirement, and then the greater value of $(T/W)_{TO}$ or $(T/W)_{climb}$ is used to size the engines. The fuel weight is derived from computations that assume a climb at best lift-to-drag ratio to a specified altitude and reserves for 45 minutes of cruise. The average specific fuel consumptions for climb and cruise are input. The equations are:

Lift-to-Drag Ratio – Maximum

$$(C_L/C_D)_{max} = 0.5\left[\frac{\pi(e)(AR)}{C_{D_o}}\right]^{0.5}$$

Thrust-to-Weight Ratio – Climb

$$(T/W)_{climb} = \frac{0.085 + \frac{1}{(C_L/C_D)_{max}}}{\sigma_{cl}}$$

Climb Speed

$$V_{cl} = 12.16 \left[\frac{(W/S)_{TO}}{\sigma_{cl} (C_L/C_D)_{max} (C_{D_o})} \right]^{0.5}$$

Climb Distance

$$D_{cl} = \frac{Alt}{0.122 \ (6080)}$$

Climb Time

$$t_{cl} = \frac{D_{cl}}{V_{cl}}$$

Fuel Weight – Climb

$$W_{fuel_{cl}} = W_{TO} \ (T/W)_{climb} \ (SFC_{cl}) \ (t_{cl})$$

Fuel Weight – Reserve

$$W_{fuel_{rs}} = SFC_{cr} \ (0.3) \ (T/W)_{TO} \ (0.75)(W_{land})$$

Data computed for cruise conditions include aircraft weight, thrust, speed, lift and drag coefficients, and fuel weight. The fuel weight for the cruise condition is derived for a calculated thrust and an input value of cruise specific fuel consumption. The equations used to develop cruise data are:

Aircraft Cruise Weight

$$W_{cruise} = W_{TO} - W_{fuel_{cl}} - \frac{W_{fuel_{cr}}}{2}$$

Cruise Thrust

$$T_{cr} = 0.3\,(W_{cruise})\,(T/W)_G$$

Cruise Speed

$$V_{cr} = \frac{1}{1.687}\left[\frac{T_{cr} + \left((T_{cr})^2 - \dfrac{4\,(C_{D_o})\,(W_{cruise})^2}{\pi\,(e)\,(AR)}\right)^{0.5}}{\sigma_{cr}\,(\rho_o)\,(C_{D_o})\,(S_W)}\right]^{0.5}$$

Cruise Speed – Maximum

$$V_{cr_{max}} = 0.8\left[29\,(518.7 - 0.00357\,(ALT))^{0.5}\right]$$

Coefficient of Lift – Cruise

$$C_{L_{cr}} = \frac{295\,(W/S)_{TO}}{\sigma_{cr}\,(V_{cr})^2}$$

Coefficient of Drag – Cruise

$$C_{D_{cr}} = C_{D_o} + \frac{(C_{L_{cr}})^2}{\pi\,(e)\,(AR)}$$

Lift-to-Drag Ratio – Cruise

$$(C_L/C_D)_{cr} = \frac{C_{L_{cr}}}{C_{D_{cr}}}$$

Fuel Weight – Cruise

$$W_{fuel_{cr}} = (W_{zero} + W_{fuel_{cl}} + W_{fuel_{rs}})\,(10^{X-1})$$

where

$$X = \frac{Range - D_{cl}}{2.3\,(C_L/C_D)_{cr}\,(V_{cr}/SFC_{cr})}$$

Data developed for design conditions include the total fuel weight, design wing loading, and design thrust-to-weight ratio. The equations are:

Total Fuel Weight

$$W_{fuel} = W_{fuel_{cl}} + W_{fuel_{cr}} + W_{fuel_{rs}}$$

Design Wing Loading

$$(W/S)_{Des} = (W/S)_{TO}$$

Design Thrust-to-Weight Ratio

$$(T/W)_{Des} = 1.05\ (T/W)_{TO}$$

The loads data required for the weight prediction equations consists primarily of gust load factors. Approximations for this data are derived in the performance subroutine, and then updated and expanded in the loads portion of the geometry subroutine. The gust load factor is derived from the wing geometry and the cruise speed. The wing is assumed rigid and the effective lift curve slope is approximated as a function of the wing aspect ratio and horizontal tail area. The equations are:

Wing Normal Force Curve Slope

$$C_{N\alpha} = \left[\frac{6}{1 + \frac{6}{\pi(AR)}}\right] + 2.15 \left[\frac{S_H}{S_W}\right]$$

Aircraft Inertia Factor — Gust Alleviation

$$U = \frac{2\ (W/S)_{Des}}{0.0765\ (\sigma_{cr})\ (C_{N\alpha})\ (\bar{C}_W)}$$

Gust Response Factor

$$K = 0.88\ \frac{U}{U + 5.3}$$

Incremental Gust Load Factor

$$\Delta N_Z = \frac{0.2006\,(V_{cr})\,(C_{N\alpha})\,(\sigma_{cr})^{0.5}\,(K)}{w\,(W/S)_{Des}}$$

Ultimate Gust Load Factor

$$N_{ZG} = 1.5\,(1 + \Delta N_Z)$$

2.1.5 BALANCE ANALYSIS. For the purpose of organizing the various aircraft weight items for balance calculations, each is placed into one of two major categories:

a. Wing and Contents

b. Body and Contents

Table 2-2 illustrates the grouping of the various aircraft items into these categories.

Table 2-2. Grouping of Aircraft Items for Balance Purposes

WING AND CONTENTS	
WING	THRUST REVERSERS FOR WING ENGINES
HORIZONTAL TAIL/CANARD	INLETS FOR WING ENGINES
VERTICAL TAIL	EXHAUST SYSTEM FOR WING ENGINES
MAIN LANDING GEAR	COOLING SYSTEM FOR WING ENGINES
USABLE FUEL IN WING	LUBRICATION SYSTEM FOR WING ENGINES
ENGINES ON WING	STARTING SYSTEM FOR WING ENGINES
NACELLE FOR WING ENGINE	OIL SYSTEM FOR WING ENGINES
BODY AND CONTENTS	
BODY	ENGINE(S) ON BODY
NOSE LANDING GEAR	NACELLE FOR BODY ENGINE(S)
SURFACE CONTROLS	THRUST REVERSERS FOR BODY ENGINE(S)
ENGINE CONTROLS	INLETS FOR BODY ENGINE(S)
FUEL SYSTEM	EXHAUST SYSTEM FOR BODY ENGINE(S)
AUX. POWER UNIT	COOLING SYSTEM FOR BODY ENGINE(S)
INSTRUMENTS	LUBRICATION SYSTEM FOR BODY ENGINE(S)
HYDRAULIC/PNEUMATIC SYSTEM	STARTING SYSTEM FOR BODY ENGINE(S)
ELECTRICAL SYSTEM	OIL SYSTEM FOR BODY ENGINE(S)
AVIONICS SYSTEM	CREW
FURNISHINGS	OXYGEN
AIR COND/ANTI-ICE SYSTEM	SURVIVAL GEAR
AUX. GEAR	GUNS
ARMAMENT SYSTEM	PYLONS
USABLE FUEL IN BODY	MISCELLANEOUS ITEMS
UNUSABLE FUEL	MISSION PAYLOAD

Balance coefficients for the aircraft items grouped under wing and contents have as their primary reference parameters:

a. Actual length of the wing, horizontal tail/canard, or vertical tail MAC

b. Relative location of the wing quarter-chord

c. Relative location of the wing engines cg

Balance coefficients for the aircraft items grouped under body and contents have as their primary reference parameters:

a. Actual length of body

b. Relative location of the body engines cg

Table 2-3 illustrates the aircraft items, the balance coefficient associated with each, and the parameters defining the coefficients.

The input coefficient values associated with the aircraft items are selected either from the user's prior knowledge of similar aircraft or from historical data, compiled from weight and balance manuals. Appendix 1 of Reference 7 lists example values of these balance input coefficients for various transports, bombers, and fighters.

Within the balance program itself, cg locations and moments of individual aircraft items are computed by use of two general equations:

a. Item cg location = item coefficient * item parameter

b. Item cg moment = item cg location * item weight

Total aircraft center of gravity station location is determined from the following equation:

$$CG_T = \frac{(CG_{.25\bar{c}} + CG_{wac})\, W_{wac} + CG_{bac}\, W_{bac}}{W_{wac} + W_{bac}}$$

Total aircraft cg location is expressed as a percentage of wing MAC by use of the following equation:

$$CG_{\bar{c}w} = \frac{CG_T - LEMAC}{\bar{c}w}$$

2.1.6 CG RANGE ANALYSIS. Plotting gross weight versus cg location allows visual observation of cg travel as each expendable item is loaded on an aircraft and subsequently expended. The resultant plot illustrates the most-forward and most-aft cg locations during takeoff, normal flight, and landing for the specified mission profile.

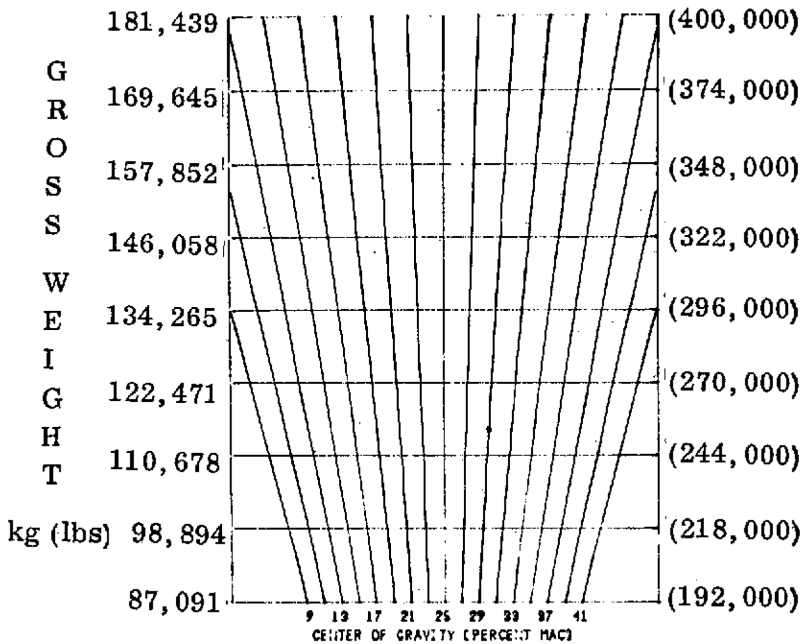
Table 2-3. Aircraft Items, Balance Coefficients, and Coefficient Definitions

WING	CWINGT	RATIO WING CG LOC AFT WING LEMAC TO WINW MAC LENG
VERT TAIL	CVT	RATIO VERT CG LOC AFT VERT LEMAC TO VERT MAC LENG
HORIZ TAIL	CHT	RATIO HORIZ CG LOC AFT HORIZ LEMAC TO HORIZ MAC LENG
MAIN LND GEAR	CMAIN	RATIO MLG CG LOC AFT WING LEMAC TO WING MAC LENG
WING ENGINES	CWENG	RATIO OF WING ENGINE CG LOC FWD OF LEMAC TO ENG LENG
WING NACELLE	CWN	RATIO WING NAC CG LOC TO WING ENG CG LOC(FWD DWMAC4)
WING REVERSERS	CWREVR	RATIO WING REV CG LOC TO WING ENG CG LOC(FWD DWMAC4)
WING INLETS	CWINLT	RATIO WING INLT CG LOC TO WING ENG CG LOC(FWD DWMAC4)
WING EXHAUST	CWEXH	RATIO WING EXH CG LOC TO WING ENG CG LOC(FWD DWMAC4)
WING COOLING	CWCOOL	RATIO WING COOL CG LOC TO WING ENG CG LOC(FWD DWMAC4)
WING LUBRICATION	CWLUB	RATIO WING LUB CG LOC TO WING ENG CG LOC(FWD DWMAC4)
WING STARTING	CWSTRT	RATIO WING STRT CG LOC TO WING ENG CG LOC(FWD DWMAC4)
BODY	CBODY	RATIO OF BODY CG LOC TO B.L.
NOSE LND GEAR	CNOSE	RATIO OF NOSE LND GEAR CG LOC TO B.L.
SURFACE CONTR.	CSC	RATIO OF SURFACE CONTR. CG LOC TO B.L.
ENGINE CONTR.	CPEC	RATIO OF ENGINE CONTR. CG LOC TO B.L.
FUEL SYSTEM	CPFLS	RATIO OF FUEL SYSTEM CG LOC TO B.L.
AUX PWR UNIT	CAXPU	RATIO OF AUX PWR UNIT CG LOC TO B.L.
INSTRUMENTS	CINST	RATIO OF INSTRUMENT CG LOC TO B.L.
HYDR/PNEUM	CHYD	RATIO OF HYDR/PNEUM CG LOC TO B.L.
ELECTRICAL	CELEC	RATIO OF ELECTRICAL CG LOC TO B.L.
AVIONICS	CAV	RATIO OF AVIONICS CG LOC TO B.L.
FURNISHINGS	CFURN	RATIO OF FURNISHINGS CG LOC TO B.L.
AIR COND/ANTI-ICE	CO	RATIO OF AIR COND/ANTI-ICE CG LOC TO B.L.
AUX GEAR	CAXG	RATIO OF AUX GEAR CG LOC TO B.L.
ARMAMENT	CARM	RATIO OF ARMAMENT CG LOC TO B.L.
BODY ENGINES	CBENG	RATIO OF BODY ENGINE CG LOC TO B.L.
BODY NACELLE	CBN	RATIO OF BODY NACELLE CG LOC TO BODY ENG CG LOC
BODY REVERSERS	CBREVR	RATIO OF BODY REVR CG LOC TO BODY ENG CG LOC
BODY INLETS	CBINLT	RATIO OF BODY INLET CG LOC TO BODY ENG CG LOC
BODY EXHAUST	CBEXH	RATIO OF BODY EXHAUST CG LOC TO BODY ENG CG LOC
BODY COOLING	CBCOOL	RATIO OF BODY COOLING CG LOC TO BODY ENG CG LOC
BODY LUBRICATION	CBLUB	RATIO OF BODY LUBR CG LOC TO BODY ENG CG LOC
BODY STARTING	CBSTRT	RATIO OF BODY START CG LOC TO BODY ENG CG LOC
CREW	CUCREW	RATIO OF CREW CG LOC TO B.L.
OXYGEN	CUOXY	RATIO OF OXYGEN CG LOC TO B.L.
SURVIVAL GEAR	CUSRV	RATIO OF SURVIVAL GEAR CG LOC TO B.L.
GUNS	CUGUN	RATIO OF GUNS CG LOC TO B.L.
PYLONS	CUPYL	RATIO OF PYLONS CG LOC TO B.L.
BODY ENGINE OIL	CBOIL	RATIO OF BODY ENG OIL CG LOC TO BODY ENG CG LOC
WING ENGINE OIL	CWOIL	RATIO WG ENG OIL CG LOC TO WG ENG CG LOC(FWD DWMAC4)
UNUSABLE FUEL	CUUFL	RATIO OF UNUSABLE FUEL CG LOC TO FUEL SYS CG LOC
MISC ITEMS	CUMIS	RATIO OF MISC ITEMS CG LOC TO B.L.
MISSION PAYLOAD	CMISPL	RATIO OF MISSION PAYLOAD CG LOC TO B.L.
L.G. RETR. MOTION	SWING	1=FWD 2=AFT 3=INBD
BODY ENGINES	QENGB	QUANTITY OF ENGINES ON BODY
WING ENGINES	QENGW	QUANTITY OF ENGINES ON WING
WING LEAD EDGE	DWLMAC	STA LOC OF WING LEADING EDGE
WING QUARTER-CHORD	DWMAC4	STA LOC OF WING QUARTER-CHORD

The most common method of presenting a cg range plot is on a grid with abscissa-value lines diverging as ordinate values increase. The cg is plotted along the abscissa and the gross weight along the ordinate. This type of plot allows a weight change to be presented as a weight-moment vector that retains its length and slope relationship anywhere on the grid. The diverging lines are necessary because a weight change has a smaller and smaller effect on the cg as the gross weight increases. In this manner, various load sequences for a mission can be plotted to give a quick picture of balance conditions throughout the flight. Figures 2-7 and 2-8 are illustrations of typical cg range diagrams.

VARIABLES

1. Weight Empty
2. Crew
3. Oxygen
4. Survival Gear
5. Guns
6. Pylons
7. Wing Engine Oil
8. Body Engine Oil
9. Unusable Fuel
10. Misc. Items
11. Fwd. Body Payload
12. Aft Body Payload
13. Fwd. Body Fuel
14. Aft Body Fuel
15. Wing Store Sta 1/8
16. Mid Body Payload
17. Wing Store Sta 2/7
18. Wing Store Sta 3/6
19. Wing Fuel
20. Wing Store Sta 4/5

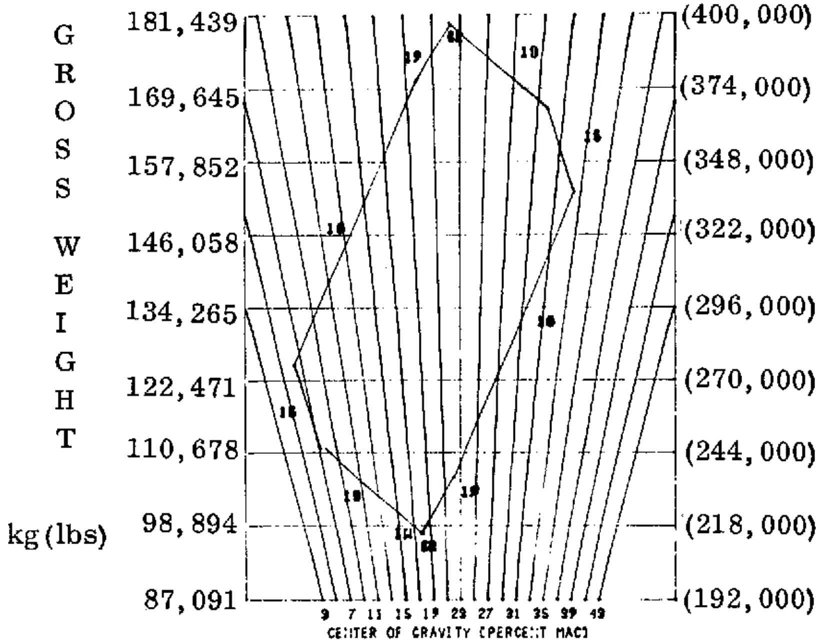


	Weight		CG	
	kg	(lbs)	cm	(in)
1.	93293	(205673)	2911	(1146)
2.	181	(400)	889	(350)
3.	0	(0)	0	(0)
4.	0	(0)	0	(0)
5.	0	(0)	0	(0)
6.	0	(0)	0	(0)
7.	43	(95)	2573	(1013)
8.	21	(47)	5438	(2141)
9.	463	(1021)	3236	(1274)
10.	3629	(8000)	716	(282)
11.	0	(0)	0	(0)
12.	0	(0)	0	(0)
13.	0	(0)	0	(0)
14.	0	(0)	0	(0)
15.	0	(0)	0	(0)
16.	36288	(80000)	2687	(1058)
17.	0	(0)	0	(0)
18.	0	(0)	0	(0)
19.	46322	(102121)	3109	(1224)
20.	0	(0)	0	(0)

Figure 2-7. Initial Display of CG Range Grid with the Variable Name, Weight, and CG Location

VARIABLES

1. Weight Empty
2. Crew
3. Oxygen
4. Survival Gear
5. Guns
6. Pylons
7. Wing Engine Oil
8. Body Engine Oil
9. Unusable Fuel
10. Misc Items
11. Fwd. Body Payload
12. Aft Body Payload
13. Fwd. Body Fuel
14. Aft Body Fuel
15. Wing Store Sta 1/8
16. Mid Body Payload
17. Wing Store Sta 2/7
18. Wing Store Sta 3/6
19. Wing Fuel
20. Wing Store Sta 4/5



	Weight kg	(lbs)	CG cm	(in)
1.	93293	(205673)	2911	(1146)
2.	181	(400)	889	(350)
3.	0	(0)	0	(0)
4.	0	(0)	0	(0)
5.	0	(0)	0	(0)
6.	0	(0)	0	(0)
7.	43	(95)	2573	(1013)
8.	21	(47)	5438	(2141)
9.	463	(1021)	3236	(1274)
10.	3629	(8000)	716	(282)
11.	0	(0)	0	(0)
12.	0	(0)	0	(0)
13.	0	(0)	0	(0)
14.	0	(0)	0	(0)
15.	0	(0)	0	(0)
16.	36288	(80000)	2687	(1058)
17.	0	(0)	0	(0)
18.	0	(0)	0	(0)
19.	46322	(102121)	3109	(1224)
20.	0	(0)	0	(0)

Figure 2-8. Plot of CG Range Diagram

Initially, the weight and cg of the basic configuration and expendable items are brought across from the weight and balance portions of the program. Changes can be effected by the operator from the graphics console. Gross weight values are obtained by adding item weights when loading the expendables and subtracting when expending them. To obtain abscissa values, cg location in percent MAC, the total moment of the basic configuration must be known along with the weight and moment arms of the load items. Ordinate values are then computed by:

$$W_T = W_1 + W_2 + \ldots\ldots + W_N$$

$$M_T = M_1 + M_2 + \ldots\ldots + M_N$$

$$M_n = W_n \times CG_n$$

$$CG = M_T / W_T$$

where

W_T = total weight

M_T = sum of individual moment arms

M_n = individual moment arm

W_n = individual weight

CG_n = individual center of gravity

CG = overall center of gravity

The following items are read into the program.

Item	What Each Includes	Where Computed
Weight	Weight of basic operating items and expendables	weight routine
cg	CG location of basic operating items and expendables	balance routine
Delta Gear Moment	Moment change due to landing gear movement	balance routine

2.2 STRUCTURAL SYNTHESIS/PARTS PREDICTION

The purpose of the structural synthesis is to utilize general preliminary design data of the type output by the vehicle synthesis to generate more detailed geometry, loads, and weight data for the primary structural components of the projected vehicle. The structural synthesis provides a means of descriptively designing structural components that fulfill specified requirements of strength and geometry. The structural synthesis process is comprised of two subprograms, one for the basic fuselage structural shell and one for the aerodynamic surface structural box. Both subprograms utilize a multi-station analysis to size the structural members. The balance of parts associated with the fuselage and aerodynamic surfaces are defined and analyzed in the part definition subprograms that are driven by the structural synthesis.

The aerodynamic surface structural box subprogram performs the following functions: distribution of the external loads, definition of shear, bending, and torsion, and definition of rib locations. It sizes the ribs, spars, and cover panels in terms of cross-sectional areas, thicknesses, and overall dimensions, and computes the theoretical weights. The part definition routines associated with the structural box define the surface geometry in terms of minimum gages, rib type and location, flange width, fastener size, etc. A breakdown is made of major components into detail parts, and logic parameters for process listings and the cost analysis are defined. These routines also define and size the leading edge, trailing edge, and tip geometry and weights.

The fuselage basic structural shell subprogram encompasses the following processes: distribution of the external loads, computation of the shear, bending, torsion, and margins of safety, and definition of the frame spacing and general fuselage barrel geometry. It sizes the frame and cover panel material thickness, cross-sectional areas, stiffener flange areas, lengths, etc., required to drive the detail part definition subprograms. The part definition routines associated with the fuselage shell define geometry in terms of frame stations, barrel stations, frame segment perimeters, etc. A breakdown is made of major components into detail parts, and logic parameters for process listings and the cost analysis are defined. An accounting is made for the fuselage penalty items (bulkheads, windows, floors, doors, etc.) not included in the fuselage structural synthesis subprogram.

Four types of weight data are computed within the program: 1) a vehicle system group weight statement per MIL-STD-254 (ASG), and, at the detail part level: 2) a theoretical or optimum weight (THEORETL WEIGHT), 3) an actual weight (ACTUAL WEIGHT), and 4) a material purchase weight (MTL WEIGHT). Detail part weights are summed to buildup subcomponents into subassemblies, and subassemblies into major components, etc., to derive the complete airframe assembly weight.

A group weight statement for the complete vehicle system is generated within the vehicle synthesis routines. The procedure is statistically based and utilizes a series of empirical equations derived from the analysis and extrapolation of historical aircraft

weight data. This level of analysis is consistent with that normally used at the preliminary design state. The output format is taken from the weight report format specified in MIL-STD-254 (ASG).

A theoretical weight, OPWT, is generated for primary structural components within the structural synthesis routines. The theoretical weight is the weight of the basic, idealized structural element. It represents an optimum value that is based on geometry of a component sized simply for load carrying capability. Real world manufacturing and assembly constraints are not considered. Typical features not accounted for are: flanges to serve as attachment points, clearance allowances, material widths for edge distance requirements, joint load path continuity, and minimum gage.

The actual weight, ACWT, reflects the actual weight of the finished part. It is computed based on the actual geometry of the finished part, and accounts for all design, manufacturing, and assembly considerations that would normally go into producing a real part. Figure 2-9 illustrates the different concepts involved in determining the idealized or theoretical weight and the practical or actual weight. The former is based on the output from the structural synthesis routines, and the latter on the detail part definition routines.

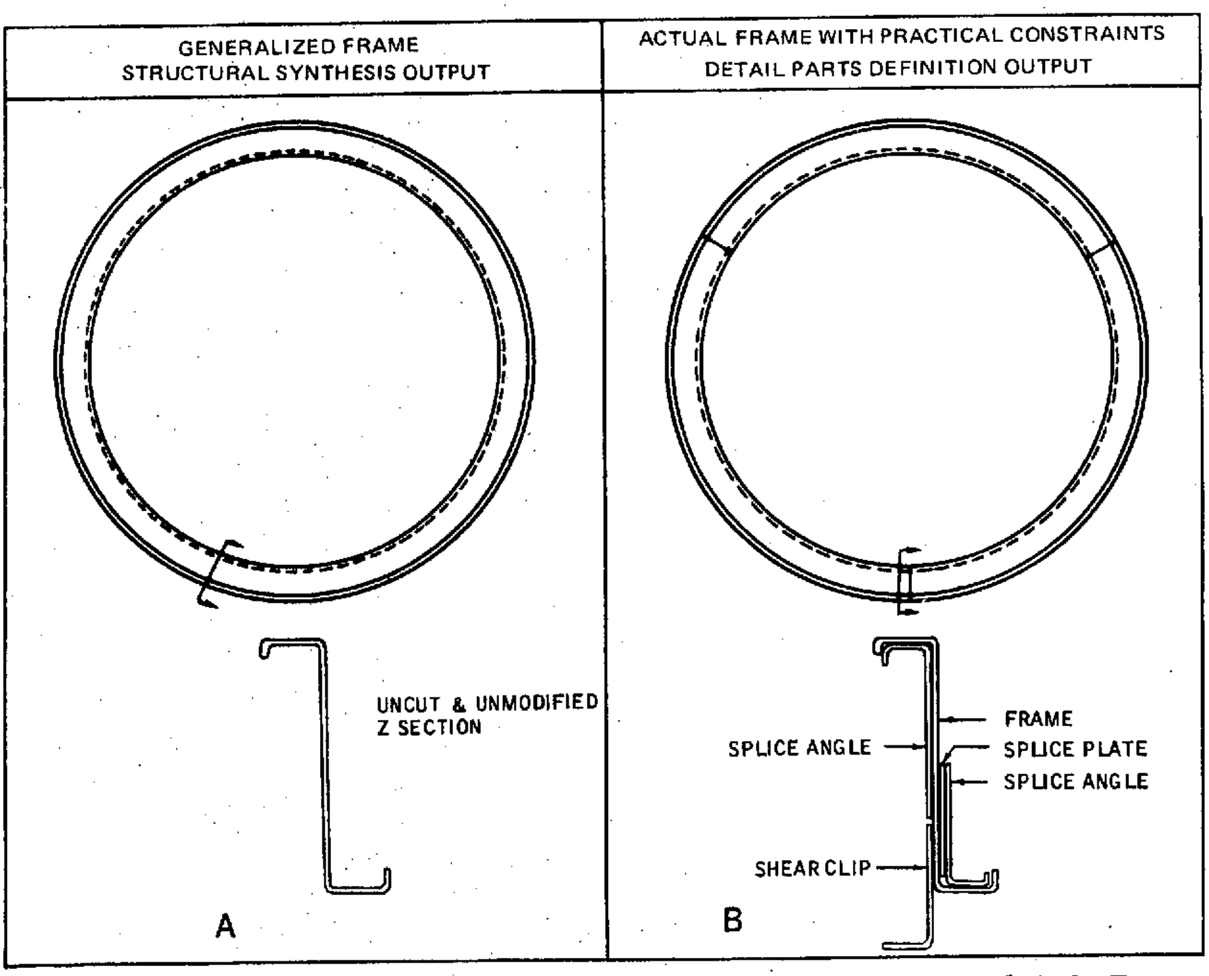


Figure 2-9. Representative Difference Between Theoretical and Actual Body Frames

The material purchase weight, MAWT, is the weight of raw material stock that must be purchased in order to be able to manufacture a part of actual weight, ACWT. Calculation of the material purchase weight uses the same terms as the actual weight but includes allowances for material removed during manufacturing. Operations resulting in the loss of material include the initial material cut off from the raw stock, initial cutting to size, trimming, milling, turning, drilling, etc. Figure 2-10 illustrates the difference in actual and material purchase weight for an integrally stiffened skin panel. Extruded plate is purchased. From the constant dimensions of the plate a skin panel with varied skin thickness and stiffener dimensions is machined corresponding to the varied load conditions over the surface of the skin. All of the required computer program inputs in the areas of the structural synthesis and parts definition utilizes the International System of Units.

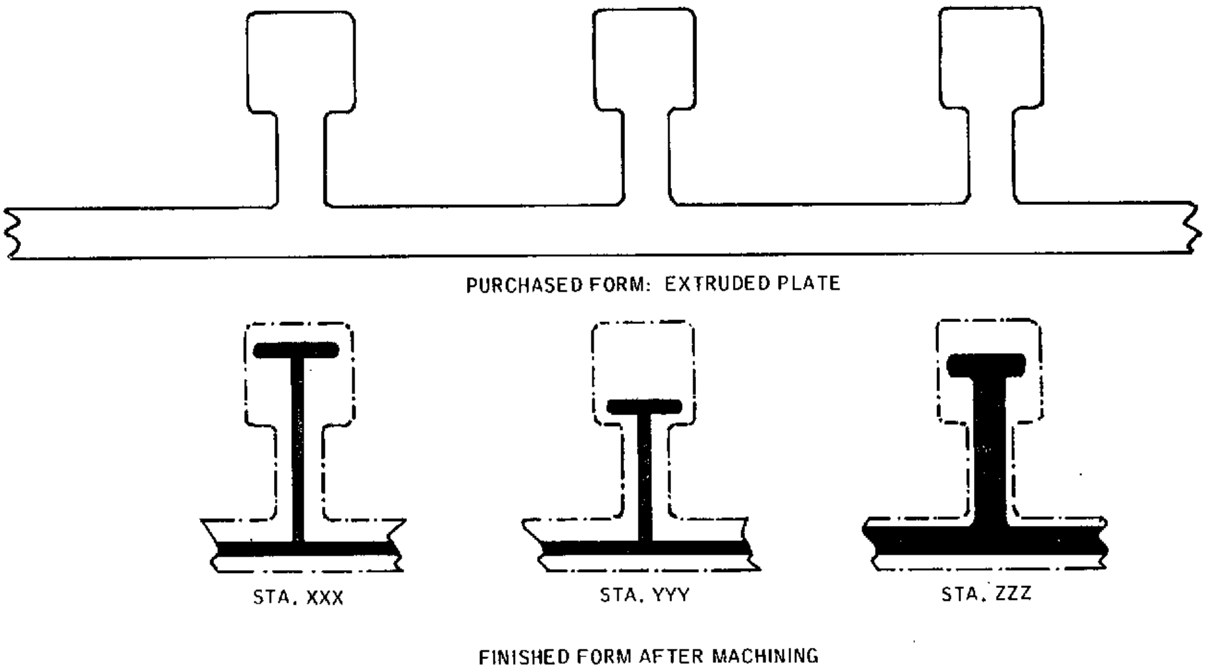


Figure 2-10. Representative Difference Between Material Purchased and Finished Form of Skin Panels

2.2.1 AERODYNAMIC SURFACES. The structural synthesis and parts definition associated with the aerodynamic surfaces is subdivided into two parts. The first part deals with the structural box and the second with the leading edge, trailing edge, and tips. The leading edge is defined to include all items located forward of the front spar, the trailing edge includes all items aft of the rear spar, and the tip includes all items outboard of the structural box tip closing rib.

The synthesis of the box structure encompasses the covers, spars, and ribs. User options include three modes of construction and eight material types. The construction modes available are skin-stringer, multi-web, and full depth sandwich. Material properties for aluminum, titanium, steel, and composites are stored in the program as a function of temperature. Loading conditions considered include a combination of

airloads due to lift and drag as well as inertial loads. Loads data for up to four conditions may be input, and a critical load envelope is computed. Output is comprised of geometry and weight data for the major components of the structural box.

The parts definition routines associated with the box structure break down the covers, spars, and ribs into their component detail parts. Actual and raw material purchase weights are computed, and the logic parameters needed for manufacturing process callout and cost analysis are defined.

The second half of the structural synthesis and parts definition process deals with the leading edge, trailing edge, and tip structures. The analysis procedures are similar to those for the box structure. The synthesis portion of the program derives the geometry for the flaps, foreflaps, ailerons, rudders, elevators, slats, spoilers, fixed leading and trailing edges, and tips. Each component is then broken down into a series of detail parts. Part weights and cost parameters are output. The structural synthesis and parts definition routines are discussed in detail in the following sections.

2.2.1.1 Structural Box Analysis. The aerodynamic surface structural synthesis subprogram, BOXSIZ, is completely documented in Reference 8 along with test case input and output samples. The approach taken in developing the routine was to visualize the steps followed by a preliminary design analyst in sizing the primary structural elements, and then to program each step, including the different design options, which are available in each engineering discipline, that drive the design process. In this way it is possible to derive an effective tool for use in optimizing the overall structure for loading, and hence for weight and cost. The approach also allows persons with backgrounds other than each of the specific engineering design specialties to aid in generating realistic design data of an early preliminary design stage.

The BOXSIZ synthesis subprogram, comprised of 15 subroutines, utilizes a multistation analysis and presently incorporates options for three modes of construction and eight material type selections. It should be noted that these options, while currently available in the aerodynamic surface structural synthesis subprogram, are not necessarily options available with respect to the remainder of the program. The primary components synthesized by the BOXSIZ subprogram are cover panels, spars, and ribs. A flow diagram of the routine is presented in Figure 2-11, and descriptions of the subroutines are given in Section 3.1 of this volume.

The three modes of construction currently available are skin-stringer (multi-rib), multi-web (multi-spar), and full-depth sandwich. Generally the type of surface (wing or tail) and the mission requirements of the vehicle will guide selection of the mode of construction. For example, transport aircraft operate at high subsonic speeds and moderate load factors. The airfoil sections have moderate thickness ratios, and the skin-stringer construction usually offers the best weight efficiency. Multi-web construction is used in the wings and tails of high performance military aircraft and is associated with high speed, high load factors, and relatively thin surfaces. Ribs in

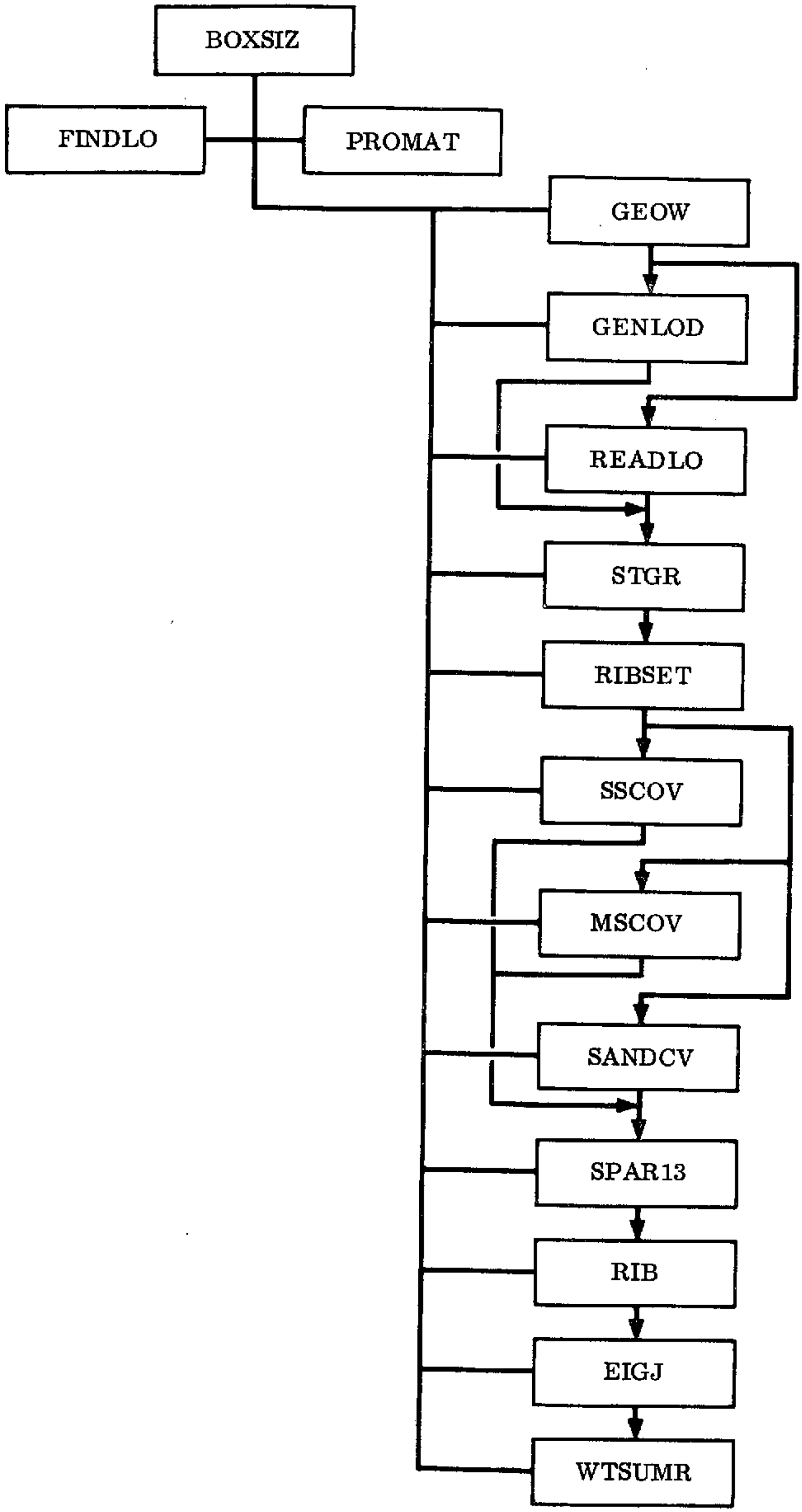


Figure 2-11. Flow Diagram of the Lifting Surface Structural Synthesis Routine BOXSIZ

this mode are usually located for a specific purpose, such as backing up an external store hardpoint or control surface hinge. Full-depth sandwich structures are most likely to be found on very thin high-speed surfaces. In cases where the selection of mode is not obvious, it is suggested that the synthesis routine for several modes be exercised and the results compared.

Skin-stringer construction, also referred to as multi-rib construction, uses closely spaced stiffening elements (ribs) and integral or attached stringers to support the skin and raise the buckling stress of the cover panel to the crippling stress of the stringer. The ribs serve to distribute airload pressures and concentrated loads, and to resist crushing due to bending. Spars carry vertical shear loads and enclose the section to form a torsion-resistant box.

Multi-web construction, also referred to as multi-spar construction, is characterized by relatively thick cover panels supported by several spanwise web (or spar) elements. Cover panels are not permitted to buckle and are usually stressed to their ultimate allowables. Ribs are widely spaced and serve to introduce concentrated loads into the box. The spanwise web elements carry vertical shear and form torsional cells with the covers.

Full-depth sandwich construction uses a core of low density material to stabilize and support the cover panels and spar webs. The core is assumed to perform the function of ribs, distributing shear and crushing loads in addition to its stabilization function. Spanwise shear is carried by front and rear spar webs.

Material properties, including density, elastic and shear moduli, and allowable tensile, compressive, and shear strengths are stored in the program as a function of temperature. The material type is input for the various elements along with the associated temperature environment. A separate material type and thermal environment may be specified for upper and lower cover panels, spars, and ribs. The eight materials currently available as BOXSIZ structural synthesis options are:

a. Aluminum alloy 2024-T6

b. Aluminum alloy 2024-58S1

c. Aluminum alloy 2219-T78

d. Aluminum alloy 7075-T651

e. Titanium alloy Ti-8Al-Mo-IV single annealed

f. Titanium alloy Ti-8Al-Mo-IV duplex annealed

g. René 41

h. Boron-epoxy composite

Primary components – cover panels, spars, and ribs – are sized in a manner consistent with standard preliminary design techniques, utilizing a load analysis as the basis. Following is a description of the primary components with a subsequent discussion of the load analysis.

Cover Panels. Three types of cover panels are presently required by the optional modes of construction. Skin-stringer covers are assumed to consist of a sheet with spanwise taper and either joined or integral stringers. Possible stringer shapes include Z, J, and hat sections for the jointed skin-stringer combination, and blade, Z, and T sections for the integral skin-stringer. These sections are summarized in Figure 2-12. Skin-stringer compression covers are treated as wide columns whose length is equal to the rib pitch. Compression covers are sized for simultaneous local and general instability. Tension covers are sized to the tensile ultimate stress of the material and checked for compressive stress under the negative bending condition.

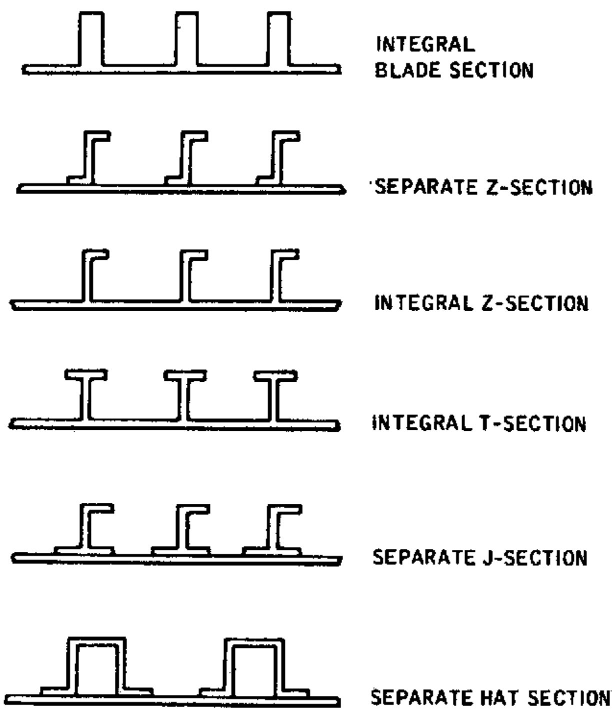


Figure 2-12. Skin Stringer Cover Panel Combinations Available in the Structural Synthesis Subprogram BOXSIZ

Multi-web cover panels may be machined from thick plate with complex integral tapers, lands, and reinforcements. Compression covers for the multi-web mode are idealized as infinitely long plates whose width is equal to the spar spacing. An intermediate edge constraint is selected representing greater support than simply supported edges, but less than the condition of sides fixed with ends simply supported.

Full depth sandwich covers are assumed to be machined from plate or extruded plate stock, and usually incorporate a considerable amount of detail and taper. Cover panels for the full depth sandwich are assumed to be fully supported by the core and are sized for maximum stress levels.

Spars. Spars are classified as exterior (front and rear) or interior, and basically are comprised of caps and web or truss elements. Six spar types are presently available:

a. Corrugated web
b. Integral web
c. Built-up web
d. Integral truss
e. Built-up truss
f. Flat sheet

Integral is used to denote machined or formed from a homogeneous piece of material. Built-up denotes an assembly of elements usually mechanically joined. Figure 2-13 illustrates some examples of available spar concepts. Spar caps are included in stiffness calculations but are neglected in bending since they contribute a relatively small area. Cover panels are assumed to provide all the bending material, and spar caps supply only the material necessary for attachments and for web shear transfer. The spar web or truss equivalent is designed to carry shear and support a compressive load. Corrugated webs provide their own stiffening by virtue of their corrugations.

Integral and built-up webs are similar in concept with stiffening elements on the web. The integral web has superior edge support characteristics and hence fewer overlaps, while a build-up web offers superior mechanical properties. The integral and built-up truss types are likewise similar in concept. The integral truss web offers more rigid joints, and the built-up truss offers better mechanical properties. Flat sheet webs lack stiffening elements and are restricted to full depth sandwich modes of construction.

<u>Ribs.</u> Rib construction is basically the same as that of spars, comprised of caps and webs or truss elements. Figure 2-14 illustrates some examples of rib concepts. Rib

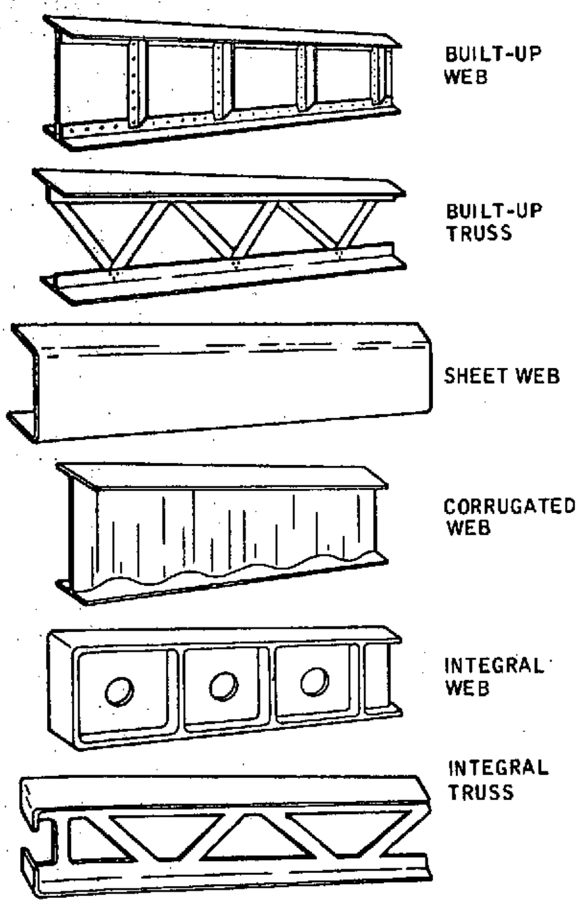


Figure 2-13. Examples of Spar Construction Types Available in the Structural Synthesis Subprogram BOXSIZ

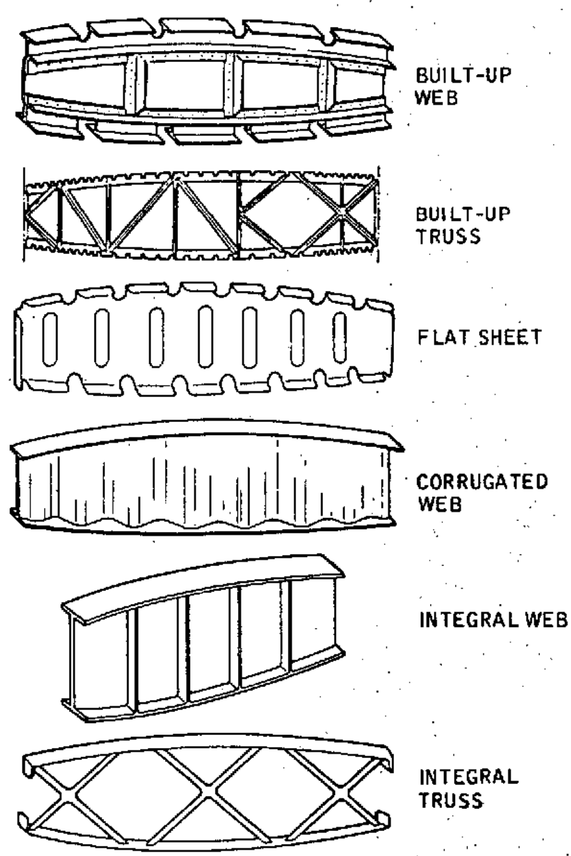


Figure 2-14. Examples of Rib Construction Types Available in the Structural Synthesis Subprogram BOXSIZ

caps are sized to react a moment at the rear due to the loading on the surface aft of the rear spar. Rib webs are sized to carry shear and to support a compressive load. The types of rib webs currently available are similar to the spar web configurations, except for the lack of a flat sheet web construction.

In general the primary loading conditions result from 1) a combination of airloads due to lift and drag, and 2) inertial loads. The minimum requirement of the synthesized structure is that it support these external loads. The usual preliminary design practice is to estimate structural sizes on the basis of these loads and to subsequently refine the sizing during development stages as additional criteria and data are refined.

Load data in the form of net shear, moment, and torsion at 10% span locations for four different conditions may be input. An envelope of critical loads is generated and used in the sizing process. Alternatively, a net load and distribution parameter such as the chordwise and spanwise center of pressure location may be utilized. The net load is given a trapizoidal spanwise distribution, which is integrated to provide shears and moments. Concentrated loads may be included with the net load option. However, the existing treatment of concentrated loads involves combining them with the overall net load distribution.

Another feature of the net load option, which is intended to improve the load representation of wings, is the inclusion of inertia relief due to fuel and structural weights, and the computation of a gust load factor (based on methods of MIL-8861). These items are not applicable to tails and are suppressed when tail surfaces are under consideration.

The internal load distribution is a function of the particular mode of construction. A rigorous analysis requires a knowledge of stiffness and mass characteristics and is most likely a redundant solution process. This level of detail is beyond the scope of a preliminary design tool; consequently, the following assumptions have been made to form the basis of the internal load analysis. The cover panels provide bending material; torsion is carried by a one-cell box consisting of upper and lower cover panels and front and rear spars; net shear is distributed equally to all spars; the torsional load accompanying the maximum bending condition is assumed small; and the reduction in compressive allowables due to shear interaction is neglected. In the skin-stringer mode of construction, the ribs distribute the airload to spars and resist crushing due to bending. In the multi-spar mode of construction the spars are assumed to distribute the airload and resist crushing. In the full-depth sandwich construction, the core supports the cover panels and spar webs, distributes the airload, and resists crushing.

2.2.1.2 Structural Box Parts Definition. To be able to predict manufacturing costs based on the actual work to be performed, a complete list of required parts must be generated. A parts definition procedure was developed that calls out a list of detail parts when a structural component such as a wing spar or a body frame is specified by the structural synthesis routines. Each detail part is used, in turn, to call out a list of the associated manufacturing processes and the raw material stock necessary to produce that part.

The part list library was established in the following manner. A model component such as the structural box of the C-5A vertical stabilizer was selected and a complete part listing of the model component was obtained. Additional parts were added to those of the model component to account for variations in the mode of construction. The example components were used as a checklist to ensure that all detail parts were accounted for; they were not intended to serve as models to develop statistical part prediction factors. An example of a component part listing for a skin-stringer type of vertical stabilizer is presented in Table 2-4. Illustrated are the type and form of the data utilized to develop the part list library. The figure reflects the number of dissimilar parts and total pieces making up each component and gives the relative weights.

The parts list developed for the program is in a more generalized form than that shown in Table 2-4. Instead of having separate front, intermediate, and rear upper and lower spar caps, for example, the program defines one general spar cap.

However, the specific dimensions are determined separately for each location by the spar sizing procedure in the structural synthesis routines. In this way the actual dimensions change for various locations and types of spar caps, but the general shape remains the same. Table 2-5 is a summary of the parts buildup available in the program for aerodynamic surfaces.

The actual part prediction is done by establishing the functional dependency between the parts available on the parts list and the specified mode of construction or the structural configuration. For example, if a skin-stringer mode of construction is specified, the structural synthesis routine calls out and sizes the appropriate spars, ribs, and cover panels. The parts definition routines then specify each detail part making up these components. The form of the functional logic used to break down each component further is shown in the following example for a vertical stabilizer, which assumes the selection of a truss-type, buildup rib, a built-up web type of spar, and an integral skin-stringer skin.

A truss-type, built-up vertical stabilizer rib is basically composed of left and right (upper and lower) caps, truss-type braces, clips for skin attachment, and fasteners. An assumed angle of 45 degrees is used to calculate the number of cross braces, which is then equal to the rib chord divided by the average rib depth. The number of right-angle braces is equal to the number of 45-degree braces minus one. The number of clips required is equal to the number of skin panels specified minus one, assuming the panels overlap and that the edges of the forward and aft panels attach to spar clips instead of rib clips. The number of fasteners required for skin attachment is derived as a function of the number of rows of fasteners needed and a typical fastener spacing, for aluminum, say four times the fastener diameter. Figure 2-15 shows a root chord section of the C-5A vertical stabilizer, illustrating the construction of the rear spar, cover panels, and the lower rib. Figure 2-16 illustrates the actual truss-type rib configuration used in the C-5A vertical stabilizer compared to the truss-type rib generated functionally by the parts definition routine.

Table 2-4. Example of a Component Part Listing

AIRFRAME ELEMENTS	WEIGHT (kg)	WEIGHT (lb)	DISSIMILAR PARTS	TOTAL PIECES
Horizontal Tail	[2,643.1]	[(5,814.9)]	[2,287]	[4,528]
Basic Structure	[2,058.9]	[(4,529.6)]	[1,366]	[2,975]
Center Section	[449.0]	[(987.8)]	[99]	[163]
Upper Caps & Covers				
Front Spar Cap	3.8	(8.4)	3	3
Intermediate Spar Cap	11.5	(25.4)	3	3
Rear Spar Cap	30.4	(66.8)	3	3
Interspar Cover	41.0	(90.1)	4	4
Joints, Splices, & Fast.	12.0	(26.4)	17	42
Lower Caps & Covers				
Front Spar Cap	3.8	(8.4)	1	2
Intermediate Spar Cap	11.5	(25.8)	-	2
Rear Spar Cap	30.4	(66.8)	2	2
Interspar Cover	40.0	(87.9)	4	4
Joints, Splices, & Fast.	13.1	(28.9)	18	45
Spar Web & Stiffeners				
Intermediate Spar	6.5	(14.4)	1	1
Joints, Splices & Fast.	1.0	(2.1)	-	-
Interspace Ribs	69.0	(151.9)	21	30
Pivot Fitting Installation	170.7	(375.6)	6	6
Pitch Trim Actuator Fitting	3.9	(8.5)	16	16
Outer Section	[1,609.9]	[(3,541.8)]	[1,367]	[2,812]
Upper Caps & Covers				
Front Spar Cap	25.0	(54.9)	3	4
Rear Spar Cap	40.9	(90.0)	6	6
Interspar Cover	467.2	(1,027.8)	23	24
Joints, Splices, & Fast.	18.8	(41.4)	26	88
Lower Caps & Covers				
Front Spar Cap	25.4	(55.8)	4	4
Rear Spar Cap	39.5	(86.8)	4	6
Interspar Cover	419.3	(922.5)	12	12
Joints, Splices, & Fast.	18.5	(40.6)	46	81
Spar Web & Stiffeners				
Front Spar	56.4	(124.0)	77	118
Rear Spar	99.4	(218.7)	101	149
Joints, Splices, & Fast.	16.8	(36.9)	1	2
Interspar Ribs	167.8	(369.1)	297	754
Leading Edge				
Cover	35.6	(78.3)	21	32
Ribs	11.2	(24.7)	49	98
Auxiliary Spars	6.4	(14.0)	5	20
Joints, Splices & Fast.	5.1	(11.2)	5	10

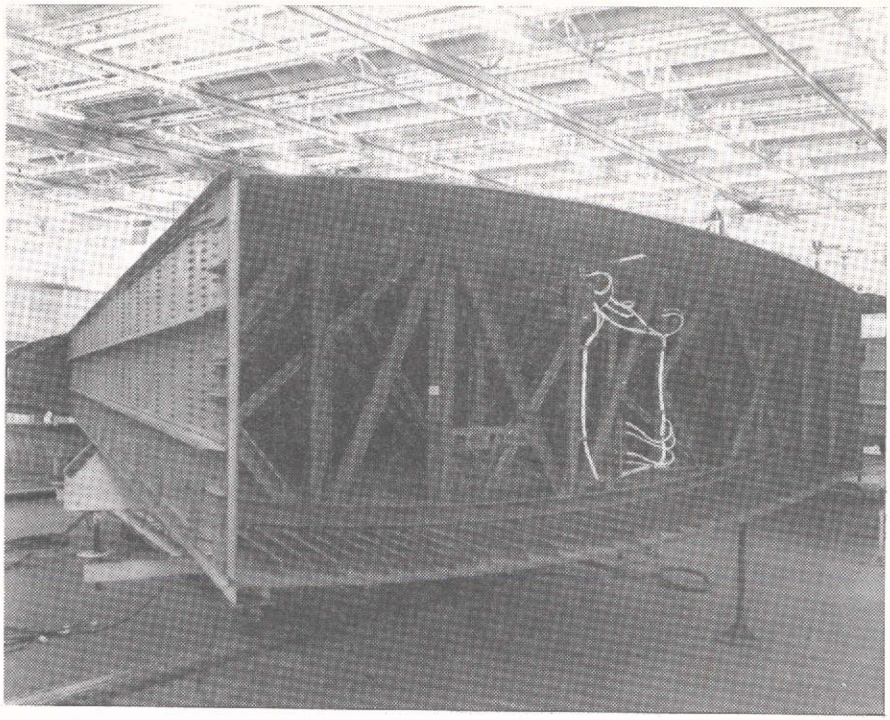


Figure 2-15. Root Chord Section of the C-5A Vertical Stabilizer

Table 2-5. Parts Summary: Aerodynamic Surfaces

- SPARS (FRONT & REAR)
  - CAPS
  - WEBS
  - RAILS
  - RIB STIFFENERS
  - WEB STIFFENERS
  - ACTUATOR STIFFENERS
  - DOUBLERS
  - DOUBLER STIFFENERS
  - ACTUATOR SUPPORTS
  - HINGE SUPPORTS
  - CLIPS
  - SHIMS
  - FASTENERS
- RIBS (STANDARDS, CLOSING, HINGE, AND ACTUATOR/HINGE)
  - CAPS
  - BRACES
  - CLIPS
  - FASTENERS
- SKIN PANELS
  - SKINS
  - FASTENERS
- FASTENERS

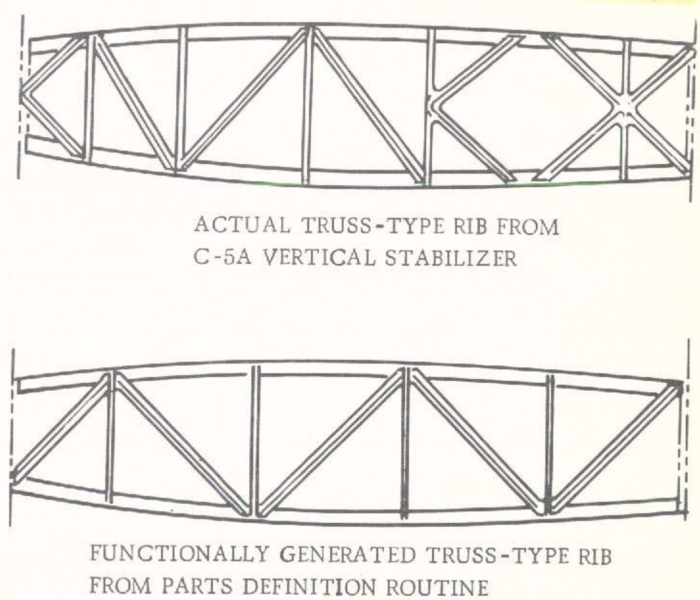


Figure 2-16. Example of an Actual Truss-Type Rib Compared to One Generated Functionally by the Parts Definition Routines

A built-up web-type vertical stabilizer spar consists of left and right (upper and lower) caps on each side of a web, which is stiffened longitudinally by left and right (upper and lower) rails. Various types of stiffeners are attached laterally across the web, including actuator and rib stiffeners plus the basic web stiffeners. Actuator and hinge support fittings and miscellaneous doublers, clips, and shims complete the parts list.

To the basic spar structure of caps, rails, and web, the program adds a rib stiffener at each rib station and one web stiffener for each rib. One doubler and doubler stiffener are required at the root of the front spar. The program also calls out two miscellaneous stiffeners and seven skin attachment clips for the front spar, and two hinge support fittings and one actuator support fitting for the rear spar. The callout for four actuator stiffeners (two each of two kinds for an actual total of six) has as its basis four actuator/hinge ribs required in a typical control surface (rudder) installation. The spar fasteners required are calculated based on the number of fasteners needed to fasten the caps, rails, and stiffeners to the web. The cap-to-web fasteners are assumed to run along the length of the spar in two rows for each cap. The rail-to-web fasteners are also assumed to run the length of the spar, but in only one row per rail instead of two. A typical spacing of four times the fastener diameter is used. The various stiffener-to-web fastener requirements are calculated based on the number of each kind of stiffener needed and the average length of each stiffener, which is assumed equal to the average spar depth. A portion of a rear spar similar to the type being discussed is shown in Figure 2-15.

The various fastener diameters used in calculations are derived on the basis of material thickness. Average thicknesses for the skin panel and spar web are calculated within the parts-listing subroutine; the skin and rib fastener diameters are based on the average skin thickness, and the spar fastener diameters are based on the average web thickness. Actual values of diameter are called from a table located in the BLOCK DATA portion of the program, which lists values of diameter corresponding to a given skin thickness.

The number of skin panels required for each side of the vertical stabilizer is calculated by assuming a maximum skin panel width of 20 inches; hence, the number of panels is equal to the rib chord divided by 20. The corresponding number of length-wise panel stiffeners is calculated based on the stiffener spacing, which is called from the structural synthesis portion of the program. The skin fasteners required are calculated as the number of fasteners needed to splice the overlapping skin panels together along their lengths using two rows of fasteners along each splice. The skin panel assembly shown in Figure 2-17 is the type discussed, and an individual panel is shown in greater detail in Figure 2-18.

The variables used in the parts definition routines, such as rib chord, average rib depth, number of skin panels, and fastener diameters, are generated as output by the structural synthesis routines and act as input for the subsequent part definition routines. There is no direct input to the parts definition routines. Three material types are currently available in the parts definition routines: aluminum, titanium, and steel. Note

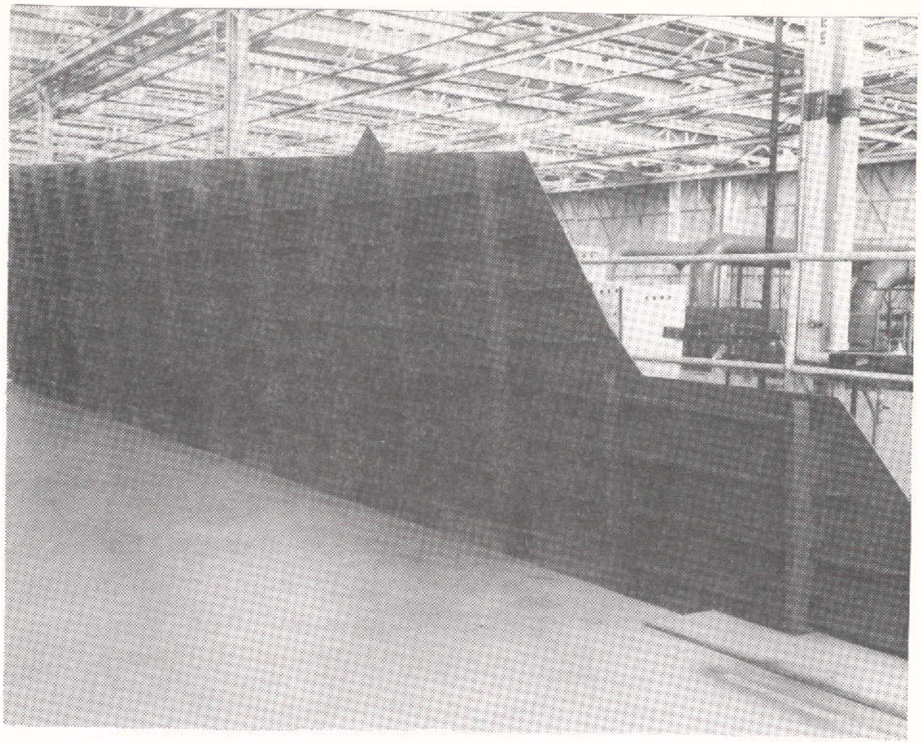


Figure 2-17. Integral Skin Stringer Panel Assembly

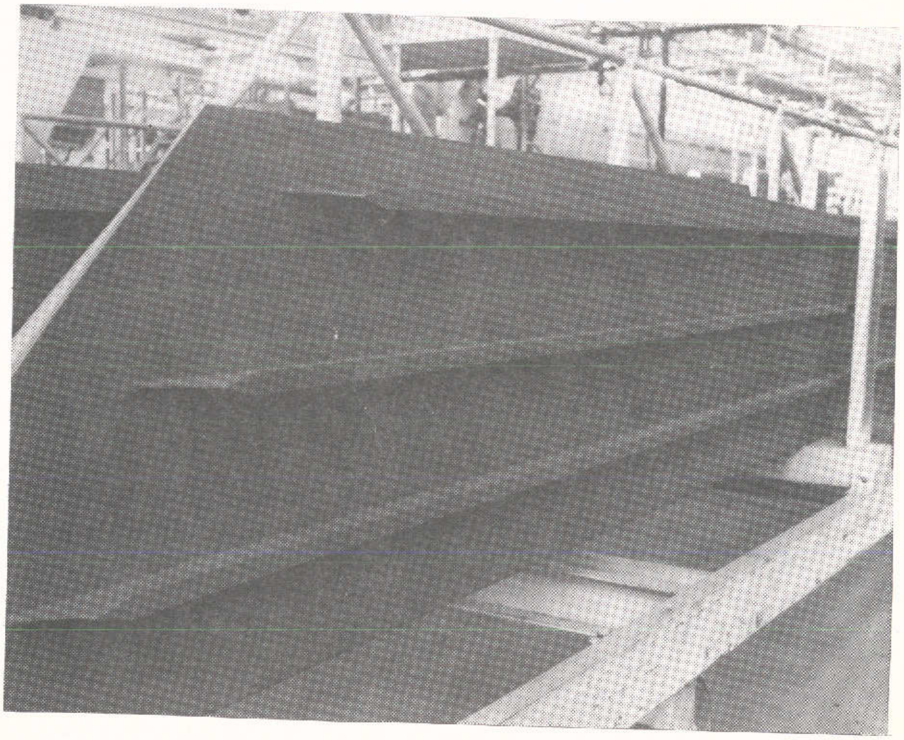


Figure 2-18. Individual Integrally Stiffened Skin Panel

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that eight material choices were available in the structural synthesis routines, including the three available in the parts definition routines. A material form is defined for all the structural components, as listed in Table 2-6. The program retains the capability of adding any number of additional material and material form choices at a future date.

Table 2-7 is a summary of some of the primary structural concepts available in the parts definition procedure. Note that the selections available in the parts definition procedure do not always correspond to the selections available in the structural synthesis routines, i.e., the number of spars presently available in the parts definition routines is two, while any number of spars may be called out in the structural synthesis routines. Provision has been designed into the program for the future addition of several alternate concepts.

Table 2-6. Summary of the Available Material Forms and the Corresponding Material Form Index

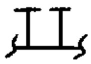
Material Form Index	Material Form	Typical Part References
1	Flat plate	Spar webs
11	 extruded plate	Cover panels
21	 extrusion	Spar caps
22	 extrusion	Spar rails
23	 extrusion	Rib caps, spar hinge/actuator supports, frames, longerons, intercostals
24	 extrusion	Rib and actuator stiffeners
25	 extrusion	Doubler stiffeners, miscellaneous stiffeners
26	 extrusion	Rib braces
27	 extrusion	Web stiffeners
44	Flat sheet	Shear clips, splice plates, ripstops, doublers, straps, spar doublers, clips, shims
81	Aluminum fastener	Fastener
82	Titanium fastener	Fastener
83	Steel fastener	Fastener

Table 2-7. Summary of Structural Concepts Available Through the Parts Definition Procedure

	Primary Mode	Alternate Mode
Spar construction	Built-up web	Integral web*
Rib construction	Built-up truss	Integral truss*
Skin construction	Integral skin-str	Built-up skin-str*
Frame construction	Built-up	Extruded*
Number of spars	Two	Input*
Number of ribs	Calc	Input
Number of lifting surface skins	Calc	None
Number of fuselage skins	Calc	Input
Number of frames	Calc	None
Number of frame segments	Calc	Input
Number of fuselage barrels	Calc	Input
Spar locations	Input	None
Rib locations	Calc	Input
Fuselage longeron spacing	Calc	Input
Fuselage frame spacing	Calc	None
Fuselage barrel lengths	Calc	Input

*Alternate mode to be added at a future date

2.2.1.3 Tip, Leading and Trailing Edge Analysis. The leading edge, trailing edge, and tip synthesis modules provide the capability to analyze the aerodynamic surface structural components that are not considered as part of the structural box. The leading edge is defined as being forward of the front spar and includes the fixed portion of the leading edge and the leading edge high lift devices (slats). The trailing edge is defined as being aft of the rear spar and includes the fixed trailing edge, foreflaps, flaps, ailerons, rudder, elevator, and spoilers. The tip is defined as that structure outboard of the structural box tip closing rib.

The synthesis includes a definition of part geometry and a detailed stress analysis that determines gages, accounts for material types, and sets minimum gage constraints. The geometry routines provide dimensional input to the stress analysis routines. The geometry and stress routines output includes part size and weight, as well as parameters for the part definition and cost routines. A generalized flow of the leading edge, trailing edge, and tip subprogram is shown in Figure 2-19.

The analysis utilizes nine geometry routines, three stress analysis routines, six supporting routines, and two calling rout ines. The geometry routines are for flaps, aileron, rudder, elevator, slat location, slats, fixed leading edge, and spoilers.

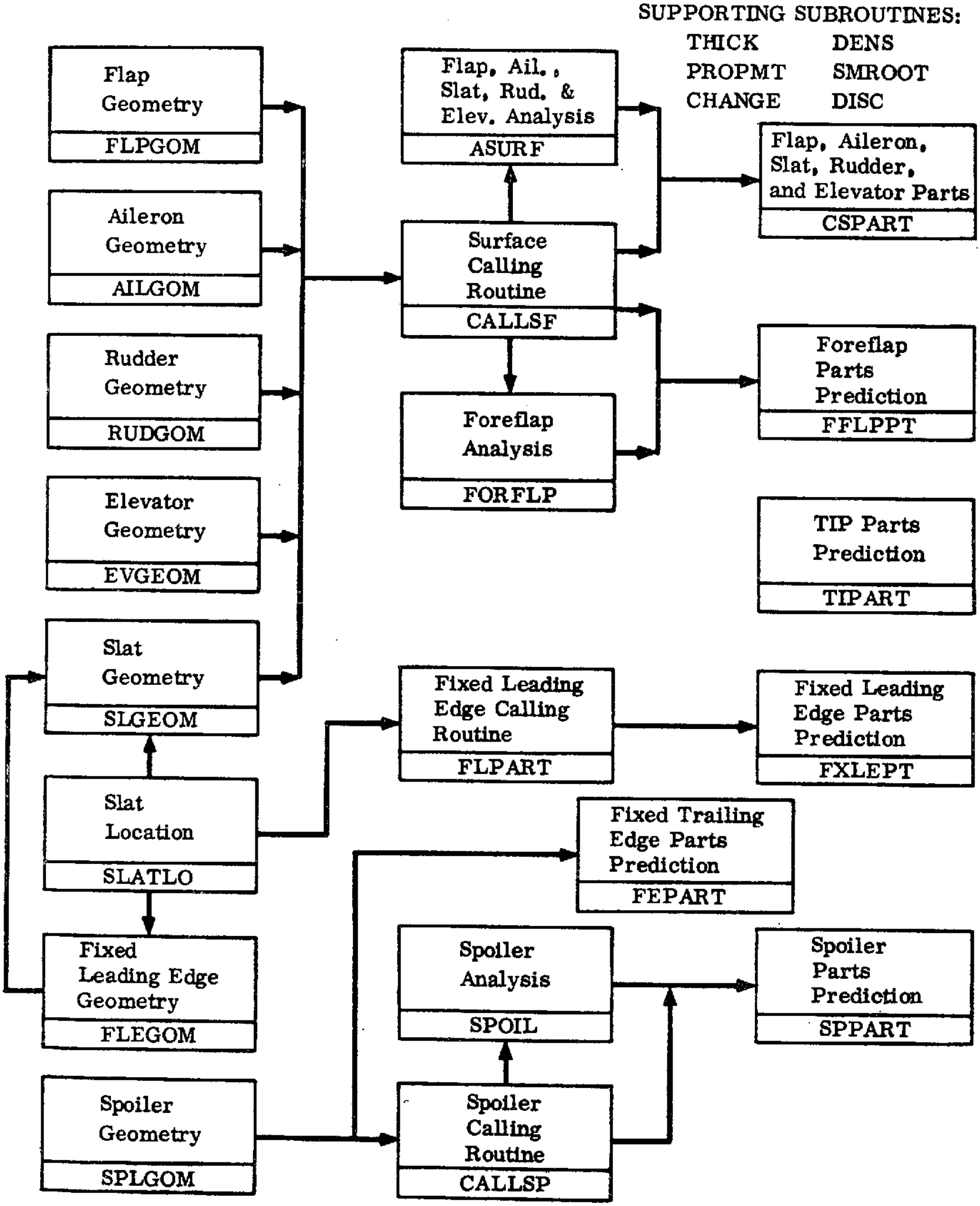


Figure 2-19. Leading Edge and Trailing Edge Synthesis Routines

The stress analysis routines include foreflap, spoiler, and one which analyzes the flaps, ailerons, slats, rudder, and elevator. The supporting routines derive dimensions, material properties, and general analysis. A discussion of these routines is included in the following paragraphs.

The flap geometry routine provides flap planform dimensions and locations from input data. The flap types considered are simple flaps, and single-slotted and double-slotted flaps. In the case of single or double slotted flaps the foreflap dimensions are computed in addition to the main flap dimensions. The driving parameters in determining flap dimensions are the flap area to wing area ratio, flap chord to wing chord ratio, and flap inboard chord. If the area ratio is input the flap length will be set to give required flap area. The flap length will be truncated at the wing tip or the inboard edge of the aileron. The flap chord is set by the ratio of flap chord to wing chord. If the ratio is zero the chord is assumed to be 85% of the distance aft of the rear spar. If the flap chord is input, the value of flap chord to wing chord ratio will be computed for use in determining flap dimensions. The inboard edge of the flap is located at the side of the fuselage. Flap geometry output consists of inboard and outboard chords, span stations of the flap inboard and outboard edges, and the flap length.

The aileron geometry routine provides aileron planform dimensions and locations from input data. The outboard edge of the aileron is assumed to be at the wing tip and the inboard edge is truncated at the side of the body if the inboard location is not specified. The aileron chord is computed as 10% greater than the trailing edge length. If the inboard edge location of the aileron is input the length will be set to provide the required aileron area. Aileron geometry output consists of inboard and outboard chords, span stations of the aileron inboard and outboard edges, and the aileron length.

The rudder geometry routine provides rudder planform dimensions and locations from input data. The rudder extends from the body to the vertical stabilizer tip. The rudder chord value is set equal to 90% of the theoretical chord length aft of the vertical stabilizer rear spar location. Rudder geometry output consists of inboard and outboard chords, span stations of the rudder inboard and outboard edges, and the rudder length.

The elevator geometry routine provides elevator planform dimensions and locations from input data. The elevator extends from the body to the horizontal stabilizer tip. The elevator chord value is set equal to 90% of the theoretical chord length aft of the horizontal stabilizer rear spar location. Elevator geometry output consists of inboard and outboard chords, span stations of the elevator inboard and outboard edges, and the elvator length.

The slat geometry routine comprises two separate operations. The first locates the inboard and outboard ends of the slats and defines the slat length. The inboard location is set at 45.7 cm (1.5 ft) outboard of the side of the body. The outboard location includes 91.4 cm (3.0 ft) of clearance for each wing mounted engine pylon. The second operation determines the individual slat lengths, chords, and inboard and outboard stations for two and four engine aircraft. The slat analysis for a two-engine configuration

provides three options for slat segment location: 1) inboard only, 2) outboard only, 3) outboard only, 4) inboard and center, 5) center and outboard, and 6) inboard, center, and outboard. The specific slat chord lengths are computed as a function of the slat chord to wing chord ratio. However, if the ratio is not input a value of 0.0735 is used. This is an average value for typical transport aircraft.

The fixed leading edge geometry routines provide planform dimensions and locations for the wing, horizontal stabilizer, and vertical stabilizer leading edges. The horizontal and vertical stabilizer leading edges start at the body and end at the tip. The leading edge chord is input as the total distance forward of the front spar. The wing has two types of fixed leading edges; under-slat and between-slat. The leading edges extend from the side of the body to the tip, the appropriate type being used as a function of the slat locations. The between-slat type extends the full distance forward of the front spar and the under-slat type assumes a chord equal to 8% of the wing chord. Fixed leading edge geometry output consists of the lengths and chords of each type of edge.

The spoiler geometry routine provides spoiler planform dimensions and locations from input data. If the spoiler area is input the spoiler will be resized to the area output from the aircraft sizing routine. If the area is not input the user must provide the inboard and outboard edge locations as well as the spoiler chord to wing chord ratio. If the spoiler chord to wing chord ratio is not input it is assumed to be 0.15. The spoiler inboard edge is assumed to be at the side of the body and the outboard edge is computed. The outboard edge is truncated at the wing tip or at the edge of the aileron. Spoiler geometry output consists of inboard and outboard chords, span stations of spoiler inboard and outboard edges, and the spoiler length.

The fixed trailing edge geometry routine assumes a total length from the body to the tip for wings, horizontal stabilizers, and vertical stabilizers. The fixed trailing edge chord is computed as a function of the total trailing edge length and the surfaces involved. The lower surface chord is computed as 6.8% of the trailing edge length if there are flaps and 10% if there are ailerons, rudders, or elevators. The upper surface chord is computed as 29.6% of the trailing edge length for flaps only. It is set equal to the spoiler chord if there are flaps and spoilers, and equal to 10% of the trailing edge length for ailerons, rudder, or elevators. If there are no control surfaces the fixed trailing edge extends from the rear spar to the aft edge of the wing, horizontal stabilizer, or vertical stabilizer.

The spoiler analysis produces structural member thicknesses and desired rivet patterns. The planform geometry is obtained from the spoiler geometry output. Member thicknesses are computed and adjusted to standard gages. Cross-sectional geometry is shown in Figure 2-20. The front spar is a bent-up sheet metal zee, the two ribs (at each support) are bent-up sheet, and the skins are sheet metal over a full depth honeycomb core.

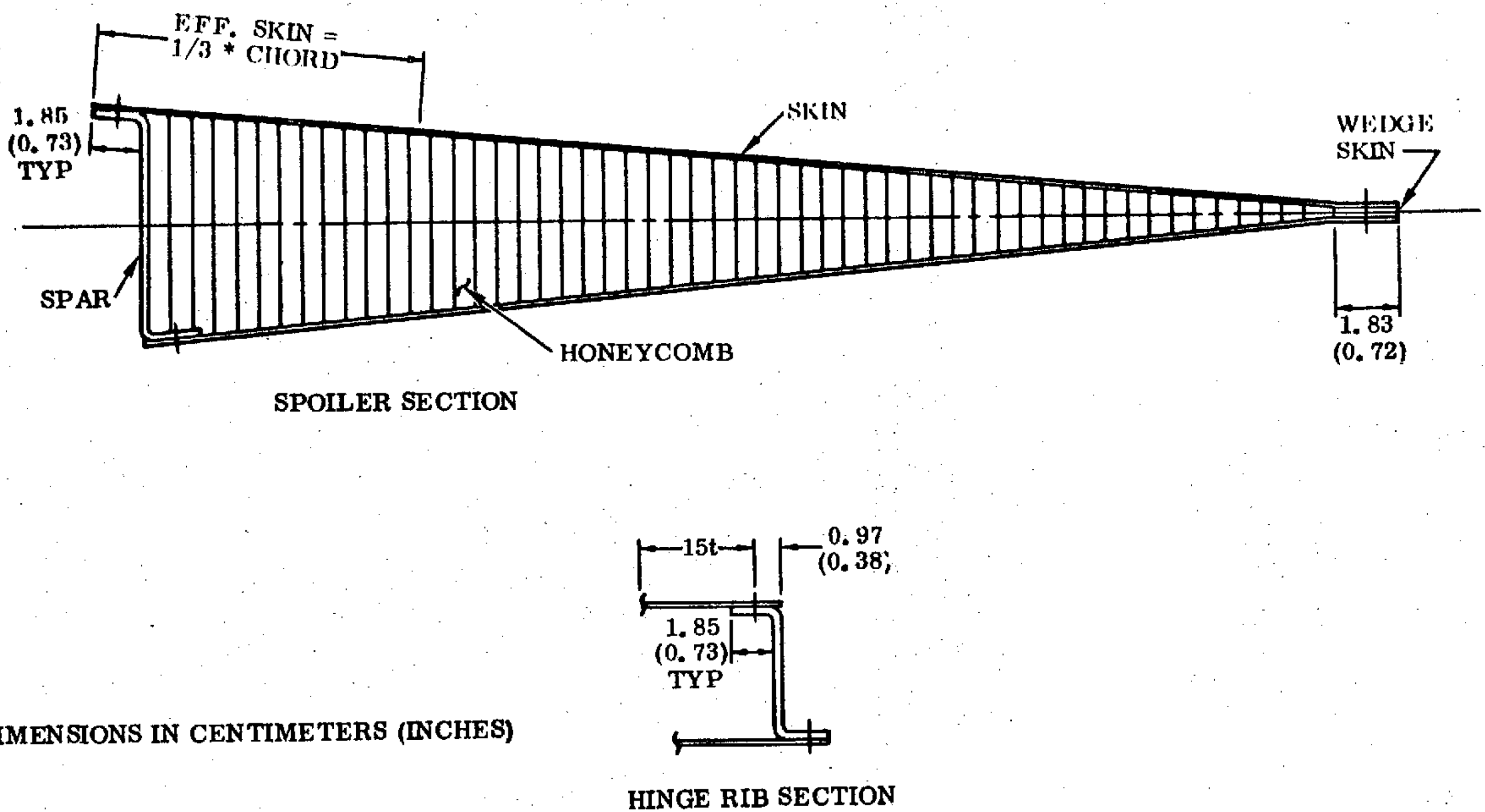


Figure 2-20. Spoiler Geometry

The spoiler analysis accounts for external and internal loads. The external loads for transport aircraft are normally those loads which the spoiler actuator produces. In this analysis the spoiler external load is assumed to be 68 N-m (600 lb-in) of hinge torque per running inch, limit. This is comparable to the 990 loading condition. The internal load analysis subdivides the total spoiler area into the smallest number of segments (individual surfaces) where all segments are equal in length and not longer than 152 centimeters (60 inches). The segments are supported at each end and all torque is taken by the inboard support. The spoiler is analyzed as a simple beam. The point of maximum bending moment is determined, and the bending moment and spar depth computed. All spoiler bending moment is taken by the spar and effective skin. The bending section (Figure 2-20) is assumed symmetrical, and the tension and compression stresses are equal to:

$$F = \frac{M\,(d/2)}{I} \qquad (2\text{-}1)$$

where

F = bending stress

d = contour depth at spar

M = bending moment

I = section moment of inertia

The compression buckling allowable is

$$F_{cs} = 0.56\, F_{cy} \left[\left(\frac{2\,t^2}{A}\right)\left(\frac{E}{F_{cy}}\right)^{0.5}\right]^{0.85} \quad \text{(Reference 9, Equation C7.4)} \qquad (2\text{-}2)$$

where

F_{cs} = compression buckling allowable

F_{cy} = compressive yield allowable

t = material thickness

A = cap area (= 1.73t)

E = material elastic modulus

The spar cap sheet thickness is sized so that the stress level is equal to or less than the larger of the compression buckling allowable or 80% of the ultimate tensile allowable.

The inboard rib is analyzed for bending at the front spar. Since all torque is taken at this rib, the bending moment is equal to the total spoiler torque about the spar. The

section (Figure 2-20) is symmetrical and the tension, compression, and compression buckling stresses are computed the same as shown for the spar.

The skin thickness is based on skin shear flow at the inboard hinge where all spoiler torque is reacted about the spar. Since the skin is supported by the honeycomb core, the shear allowable is based on the ultimate shear stress times a rivet factor of 0.8.

Appropriate material properties are selected for each part analyzed. The analysis determines the material thicknesses as a minimum required thicknesses and then rounds the value of the next larger standard gage. A minimum gage of 0.051 cm (0.020 in.) and a maximum gage of 0.635 cm (0.250 in.) are set as constraints. The standard sheet gages used are summarized in Table 2-8.

Table 2-8. Standard Sheet Gages

cm	in.	cm	in.
0.051	0.020†	0.180	0.071
0.064	0.025	0.203	0.080
0.081	0.032	0.229	0.090
0.091	0.036	0.254	0.100
0.102	0.040	0.318	0.125
0.114	0.045	0.406	0.160
0.127	0.050	0.483	0.190
0.160	0.063	0.635	0.250‡

† minimum ‡ maximum

The number of rivet holes (representing the actual number of rivets needed) and the hole sizes are output. The quantity and size of the rivets is based on a T/2A shear flow analysis at the inboard rib. The rivets are sized based on the protruding head shear allowables at a spacing of four times the shank diameter. The number of holes is equal to the number of rivets. That is, the holes are counted for only one member. When two rows of rivets are required, an additional amount of spar or rib cap width is output, but the additional area is not used to resize the cap.

The foreflap analysis produces the structural member dimensions and desired rivet patterns. The planform geometry is obtained from the foreflap geometry output. Member thicknesses are computed and then adjusted to standard gages. A typical foreflap cross section is shown in Figure 2-21. The front spar is bent-up sheet metal channel and is sized by a loads analysis. The leading edge skin and rib thicknesses are fixed at 0.127 cm (0.050 in.). The honeycomb box factor is set at 1 and assumes an allowable shear stress of 110 N/cm^2 (160 psi). The box skin thickness is assumed to be 0.051 cm (0.020 in.).

The foreflap spar analysis accounts for external and internal loads. The external applied loads are derived from the general formula:

$$W = S\, C_n \left(\frac{V^2}{295}\right) \tag{2-3}$$

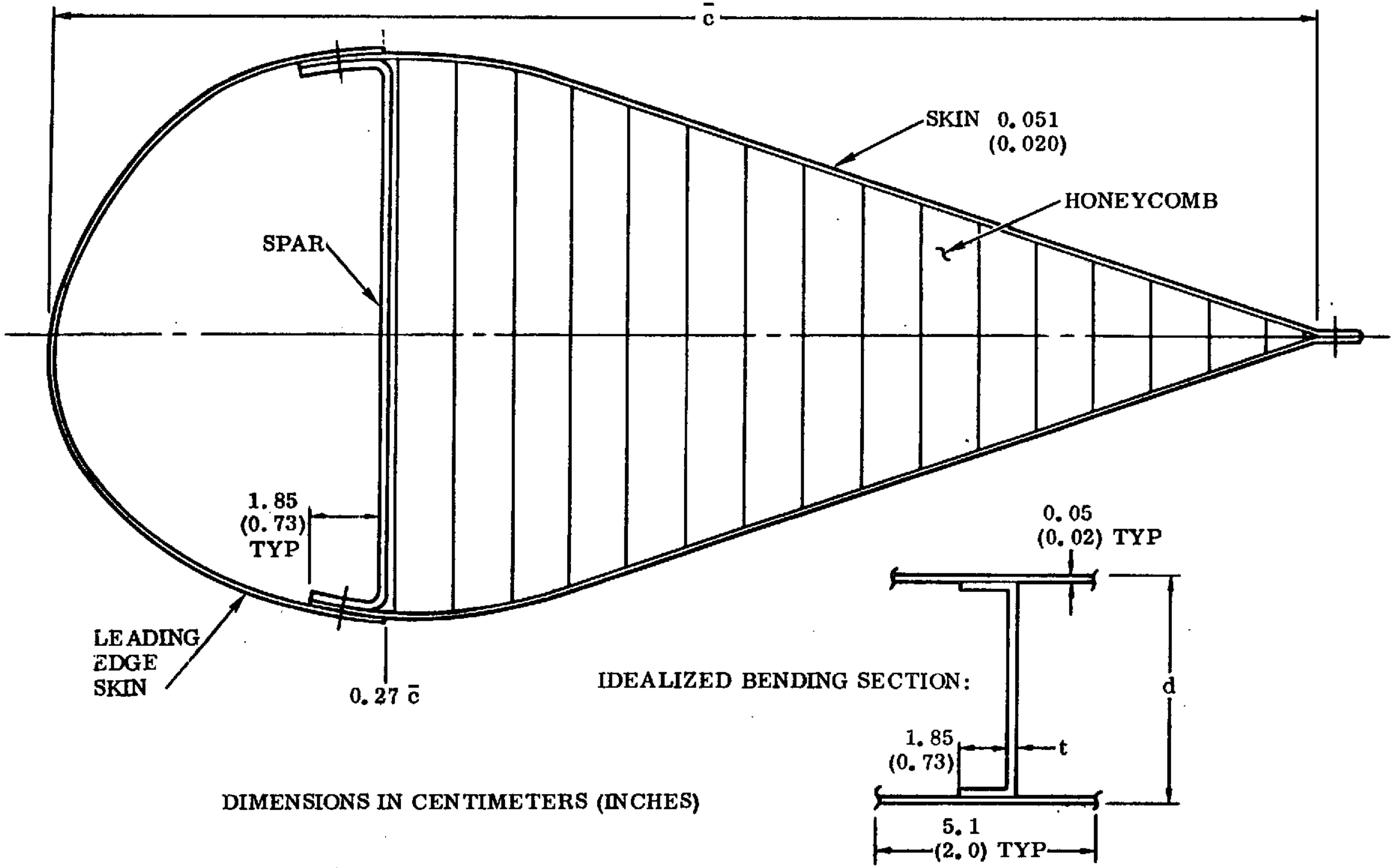


Figure 2-21. Foreflap Geometry

where

W = total surface load

S = total surface area

C_n = normal lift coefficient

V = design speed

The average pressure, ultimate, is applied to the foreflap uniformly and is computed from the transposed form:

$$P_{ave} = \frac{W}{S} = 1.5 \left(\frac{C_n V2}{295} \right) \tag{2-4}$$

where

P_{ave} = average ultimate surface pressure and for the foreflap

C_n = 4.0

V = 1.75 V_s, where V_s = stall speed

The internal load analysis subdivides the total surface length into a number of equal length segments (individual surfaces) each with a length equal to or less than 457 cm (180 in.). If the individual segment length turns out to be greater than 356 cm (140 in.), three hinge supports are assumed. One is in the center and two are located 15% of the surface length from each end. If the individual surface length is less than or equal to 356 cm (140 in.) , two hinge supports are assumed, each 28% of the surface length from each end.

The vertical shear, bending moment, and torque about the front spar are calculated at each hinge. The torque is calculated at each end of the surface segment and is assumed to vary linearly between the ends. The torque is reacted at each hinge using the same formulae used to calculate shear reactions. The foreflap bending is assumed to be taken by the spar and associated skin as shown in Figure 2-21. The bending stress can be computed from Equation 2-1, and the compression buckling allowable stress can be computed from Equation 2-2. Spar thickness is sized to be the minimum necessary so that the stress level is equal to or less than the larger of the compression buckling allowable or 80% of the ultimate tensile allowable.

All rivet patterns are assumed to be comprised of a single row of 0.65 cm (0.25 in.) diameter rivets spaced at two diameters. The output number of holes is equal to the number of rivets. However, each rivet is accounted for in only one part of the joint. Adjustment of material thicknesses to a standard gage is accomplished in the same manner as discussed for the spoiler.

The analysis of the flaps, ailerons, slats, rudder, and elevators produces the structural member dimensions and the desired rivet patterns. The planform geometry is obtained from the specific control surface geometry output, and the member thicknesses are computed and then adjusted to standard gages. The control surfaces are assumed to have the geometry shown in Figure 2-22. The front spar has extended caps and a sheet metal web, and the rear spar is a bent-up sheet. Both the leading edge skin and the main box skin are sheet metal. The trailing edge consists of a full-depth honeycomb core with a single piece of sheet metal forming both upper and lower skins. The airload ribs and the leading edge ribs are bent-up metal. There is a leading edge rib at each airload rib span station. The hinge ribs consist of extruded spar caps and a sheet metal web with bent-up flanges to pick up front and rear spars.

Appropriate material properties are selected for the analysis of each part. Thicknesses are fixed for the leading edge skin and ribs, airload ribs, rear spar, and trailing edge skin as follows:

Part	Thickness
Leading edge skin	Same as box skin
Leading edge ribs	Same as airload ribs
Airload ribs	One gage heavier than skin
Rear spar	One gage heavier than skin
Trailing edge skin	Minimum gage

The analysis for the remaining parts determines the material thicknesses in terms of a minimum required thickness and then rounds the value to the next larger standard gage. Standard sheet gages are summarized in Table 2-8, and standard gages for extrusions in Table 2-9.

Table 2-9. Standard Extrusion Gages

cm	in.	cm	in.
0.127	0.050†	0.318	0.125
0.160	0.063	0.395	0.156
0.198	0.078	0.478	0.188
0.239	0.094	0.635	0.250‡

† minimum ‡ maximum

The parts sized by a loads analysis include the basic skins, spar webs, spar caps, hinge rib caps, hinge rib webs, and the trailing edge honeycomb. The analysis accounts for both the internal and external loading conditions. The applied external loads are normal (to the surface) loads only. For the wing surfaces (flaps, ailerons, and slats) these normal loads are derived from the general formulae of Equations 2-3 and 2-4.

For flaps,

$V = 1.75\ V_s$ (Ref. MIL-8860, Para. 6.2.3.9), where V_s = stall speed

$C_n = 1.6$

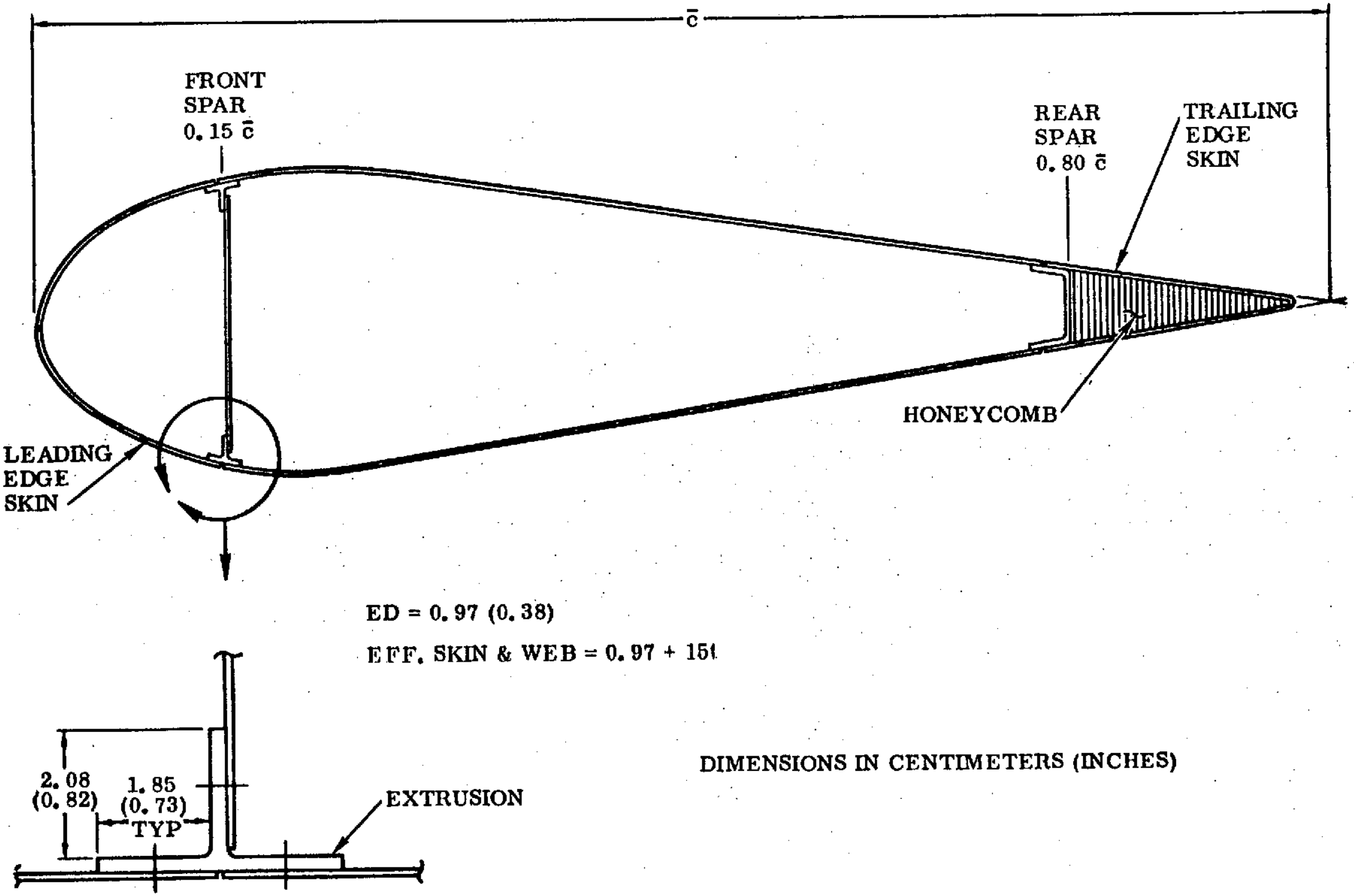


Figure 2-22. Typical Geometry for the Flaps, Slats, Ailerons, Rudder, and Elevators

For slats,

$$V = 1.75\ V_s$$

$$C_n = 3.0$$

For ailerons, rudders, and elevators, V is derived from

$$N_z W = C_{L_{Max}} S_{Wing} \frac{V_a^{\,2}}{295} \quad \text{(MIL-8860, Para. 3.2.2.2);}$$

or transposing:

$$V_a = V = \frac{295\ N_z W}{C_{L_{Max}} S_{Wing}}$$

where

N_z = maximum normal load factor

W = aircraft gross weight

$C_{L_{Max}}$ = maximum lift coefficient

S_{Wing} = wing area

V_a = aileron design speed

For ailerons,

$$C_n = 1.6$$

For rudders and elevators,

$$C_n = 1.3$$

The average pressure, P_{ave}, is applied to the control surface as a chordwise triangular distribution with the center of pressure at the 33% chord aft of the leading edge. If the design speed is equal to or greater than Mach 1, the center of pressure for the aileron, rudder, or elevator is assumed to be at the 47% surface chord. Spanwise running surface loads are therefore proportional to surface chord.

The internal load analysis subdivides the total surface length into a number of equal length segments (individual surfaces) each with a length equal to or less than 457 cm (180 in.). If the individual segment length is 356 cm (140 in.) or less, two hinge

supports are assumed, located 28% of the total length from each end. If the segments are greater than 356 cm (140 in.), three hinge supports are assumed. One is located in the center and two are located 15% of the total length from each end. The vertical shear, bending moment, and torque about the front spar are calculated at each hinge. Torque is calculated at each end of the surface segment and is assumed to vary linearly between the ends. For flaps and slats, torque is reacted at each hinge using the same formulae used to calculate shear reactions. For ailerons, rudders, and elevators all torque is reacted at the inboard (or lower) hinge.

The skin thickness is computed based on skin shear flow, and the allowable stresses are fixed as a function of rib spacing. Since the hinge rib number and locations are fixed, rib spacing is determined for each bay between hinge ribs by equally spacing airload ribs. For a given skin thickness, rib spacing can be determined from Figure 2-23. This curve is a typical design curve for sonic fatigue requirements associated

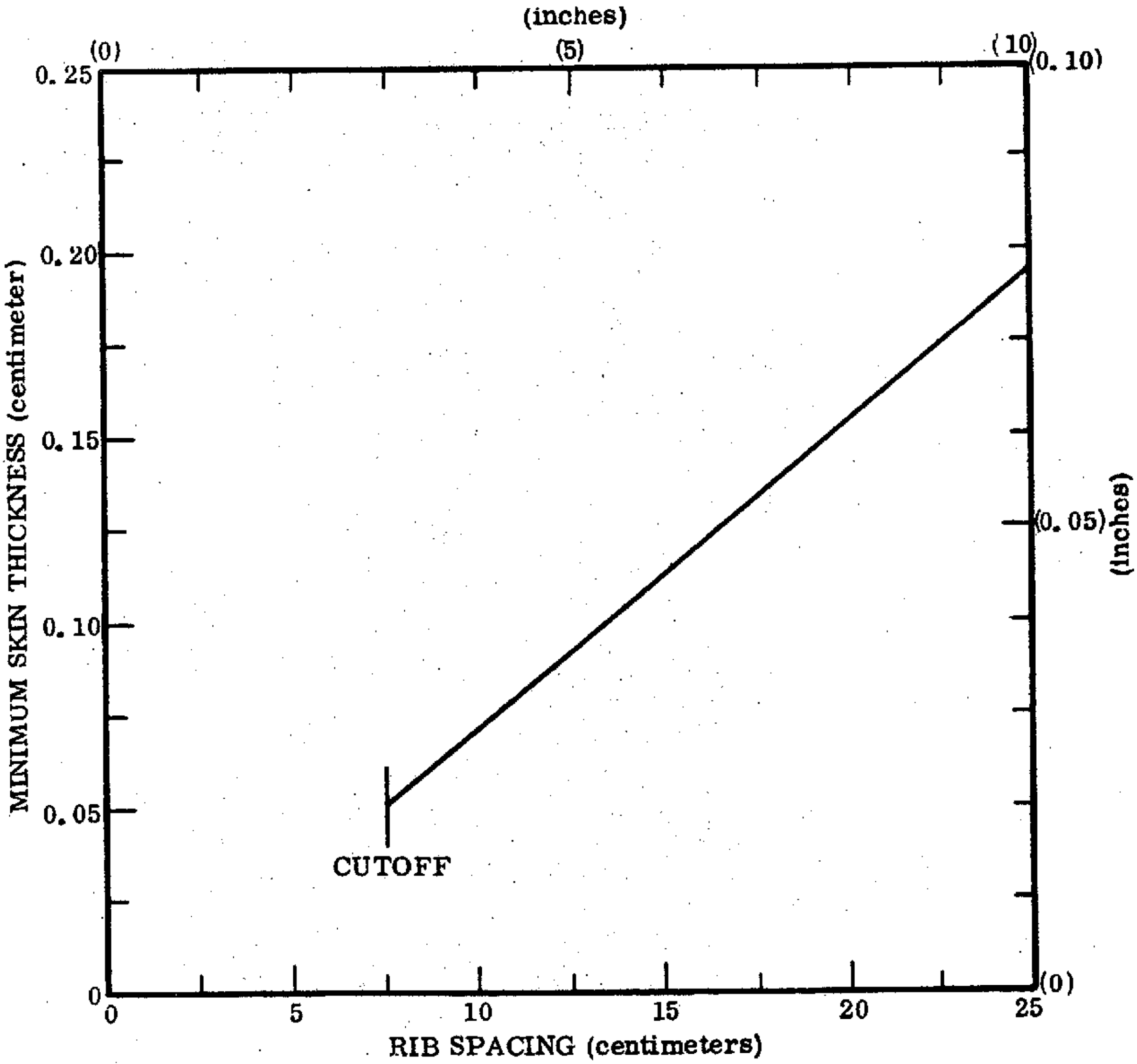


Figure 2-23. Sonic Fatigue Curve

with the preliminary design phase of aircraft analysis. For practical considerations, a minimum rib spacing of 7.6 CM (3.0 in.) has been incorporated into the computer program analysis logic.

An analysis is made of the inboard panel of the bay with the maximum rib spacing assuming maximum skin shear flow exists there. Allowables are determined for an incomplete diagonal-tension panel utilizing NACA TN 2661 (Reference 20). The critical buckling stress is computed from $F_{S_{CR}} = K_{ss}\ E\ (t/d)^2$ where K_{ss} is from figure 12 of a NACA TN 2661 (Reference 20). The diagonal tension factor, K, is derived from Equation 27 of NACA TN 2661 (Reference 20). Then the allowable shear stress can be determined as a function of K utilizing the 40-degree curve of Figure 19 (a) of NACA TN 2661 (Reference 20). The skin is sized so that the maximum shear stress does not exceed the allowable, and so that the ratio of the maximum to the critical shear stress does not exceed 5.

The spar web thickness is determined using the maximum spar shear flow. The analysis is made using either the panel at the inboard end of the surface segment or the panel just outboard of the inboard hinge, which ever has the greatest ratio of spar height to rib spacing. An incomplete diagonal-tension analysis is made like that made for the skin.

All flap bending moment is taken by the front spar caps and associated skin and spar web. The critical bending location is at the hinge where the ratio of bending moment-to-spar depth is largest. The effective spar section is as shown in Figure 2-22.

This bending section is symmetric; therefore, tension and compression stresses are equal and may be computed from Equation 2-1.

$$F = \frac{M(d/2)}{I}$$

d = contour depth at spar

The compression buckling allowable,

$$F_{cs} = 0.67\ F_{cy} \left[\left(\frac{3\,t^2}{A} \right) \left(\frac{E}{F_{cy}} \right)^{0.5} \right]^{0.40} \quad \text{(Reference 9, Equation C7.5)}$$

where

F_{cs} = compression buckling allowable

F_{cy} = compressive yield allowable

t = material thickness

A = cap area ($= 1.46\, t^2 + 0.82t$)

E = material elastic modulus

The spar cap is assumed to be an extrusion with a constant section thickness sized so that the stress level is equal to or less than the larger of the compression buckling allowable or 80% of the ultimate tensile allowable.

For all surface types, hinge ribs are assumed to have the same part thickness as the inboard hinge. The rib cap is sized by the rib bending moment at the front spar, which is equal to the surface torque (about front spar) at the inboard hinge. The generalized effective rib section is considered to be the same as the spar section. The compression buckling allowable stress equation is the same as that used for the spar. The rib cap is assumed to be an extrusion, and the constant section thickness is sized in the same manner as the spar cap. The web thickness is sized to be adequate for the inboard hinge rib shear flow,

$$Q = \frac{T}{2A}$$

Q = inboard hinge shear flow

T = torque reacted by the inboard hinge

A = inter-spar box area at the inboard hinge

The shear buckling stress is calculated for a web panel at the front spar assuming a panel aspect ratio of 2.

$$F_{SCR} = 5.9\, E \left(\frac{t}{h/2}\right)^2$$

where

F_{SCR} = shear buckling stress

E = material elastic modulus

t = material thickness

h = front spar height at rib

The web thickness is sized so that the shear stress level is equal to or less than the larger of the shear buckling stress or 80% of the ultimate shear allowable.

The assumed honeycomb type and size has a shear allowable of 110 N/cm^2 (160 lb/in^2). A factor is developed that indicates how much heavier, than the basic core, the actual core must be. The factor, K_{core}, is based on the core shear due to trailing edge airload,

$$f_s = \frac{(0.2\ P_{max})\ (0.2\ \text{chord})}{2d}$$

where

f_s = core shear

P_{max} = maximum airload

chord = chord length

d = contour height at rear spar

and

$K_{core} = f_s/160$

Rivet sizes and numbers are calculated using the shear flows that sized the skin, spar web, and hinge rib web. Rivet shear values are used as the allowables and a rivet spacing of four diameters is assumed.

Q		No. of		Spacing	
N/cm	lb/in	Rows	Rivet	cm	in.
0 to 1359	(0 to 776)	1	4AD	1.27	(0.50)
1360 to 1671	(777 to 954)	1	5AD	1.59	(0.625)
1672 to 2755	(955 to 1573)	1	6DD	1.91	(0.75)
2756 to 3427	(1574 to 1957)	2	5DD	1.59	(0.625)
3428 and above	(1908 and above)	2	6DD	1.91	(0.75)

In the output the number of holes is equal to the number of rivets; each rivet hole is accounted for in only one part of the joint. When two rows of rivets are required, an additional spar or rib cap width is output. This additional area is not used to resize the cap.

2.2.1.4 Tip, Leading and Trailing Edge Part Definition. The tip, leading edge, and trailing edge part definition routines define the detail parts making up the fixed leading edge, fixed trailing edge, slats, flaps, foreflaps, control surfaces (spoilers, ailerons, rudder, and elevators), and tips. The data that is generated includes the number of parts, part dimensions, weight, and cost parameters. The parts definition derives its input from previous geometry and analysis subroutines.

The fixed leading edge segments, as defined by the geometry subprogram, are divided into a number of 152-cm (60-in.) sections with one shorter section. If the segment is 152 cm or less, only one section is assumed. The under-slat leading edge is made of two skins spliced at the nose with an extruded angle (chafing strip). The between-slat leading edge has a one piece skin; the skin perimeter is assumed equal to 2.5 times the fixed leading edge chord. The upper skin of the under-slat segment utilizes a factor of

1.5 and the lower skin a factor of 1.0. The skin thickness is set at 0.102 cm (0.040 in.) with the chafing strips and edge member thicknesses set at 0.152 cm (0.060 in.). The ribs are spaced at 25.4-cm (10-in.) increments, and the rib height is assumed to be 0.85 times the rib chord length. The ribs are made of bent-up 0.102-cm (0.040-in.) sheet with lightening holes. The rib-to-skin fasteners are 0.40 cm (5/32-in.) diameter rivets spaced at 1.91-cm (0.75-in.) intervals. The chafing strip rivets are 0.40 cm (5/32 in.) in diameter spaced at 1.59-cm (0.625-in.) intervals, and the edge member-to-skin rivets are 0.48 cm (3/16 in.) in diameter spaced at 3.81 cm (1.5 in.).

The fixed trailing edges for the wings, horizontal stabilizer, and vertical stabilizer, illustrated in Figure 2-24, are assumed to be comprised of flat sheet skins and bent-up sheet ribs. All skins are 0.08-cm (0.037-in.) thick and, like the fixed leading edge, are defined in terms of 152-cm (60-in.) segments. The ribs are spaced at 25.4-cm (10-in.) increments and are constructed of bent-up 0.102 cm (0.040-in.) sheet with a 1.85-cm (0.73-in.) flange on each edge. Lightening holes are spaced at 3.8-cm (1.5-in.) intervals and have a diameter of 0.375 times the local chord. The skins attach along the forward edge and along each rib with 0.40-cm (5/32-in.) diameter rivets spaced at four diameters.

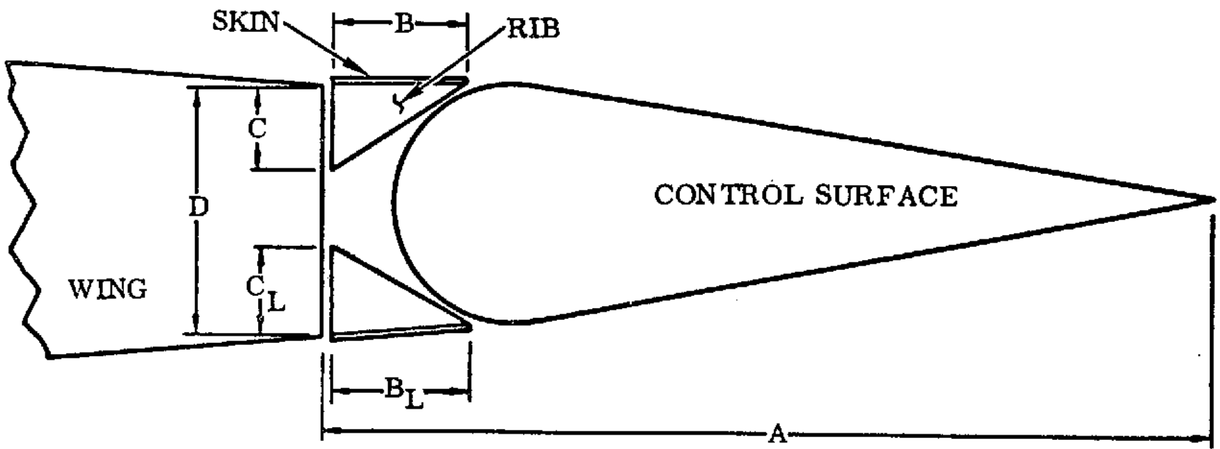
The spoiler, illustrated in Figure 2-20, is assumed to be comprised of a spar, skins, honeycomb core, and a wedge shaped skin closure. The parts definition process defines the dimensions, and the rivet sizes and quantities based on the spoiler stress analysis. The material weight assumes 2.5 cm (1.0 in.) added to the length and width dimensions of the sheet flat pattern, and to all dimensions of the full-depth honeycomb core. The material weight of the core includes 0.5 kg/m^2 (0.1 lb/ft^2) for adhesive.

The parts definition for the foreflap (Figure 2-21) derives the dimensions, and the rivet sizes and quantities from the foreflap stress analysis. The upper, lower, and leading edge skins have material weights calculated assuming 2.5 cm (1.0 in.) of additional material on all sides. The leading edge skin width, or cross-section periphery, is set equal to 2.64 times leading edge chord. Foreflap cross-sectional area aft of the spar is calculated as

$$\text{Area} = (\text{spar height}) \times (\text{chord length aft of spar}) \times 0.698$$

This formula provides the basis for computing the honeycomb core and closing rib weights. Material weight for the core is based on maximum dimensions plus 2.5 cm (1.0 in.). Closing rib material weight is based on flat pattern dimensions plus 2.5 cm (1.0 in.) on each side.

The parts definition process for the flaps, ailerons, rudders, elevators, and slats (Figure 2-22) derives the dimensions, and rivet sizes and quantities from the control surface stress analysis. The surface skins are assumed to be made in three pieces. The inboard and outboard skins are assumed to have a length equal to 28% of the surface length and the center 44% of the surface length. The leading edge skin width (periphery of leading edge cross section) is calculated from the following:



D = 0.85 (MAX. WING THICKNESS)

C = LESSOR OF B OR D/2

C_L = LESSOR OF B OR D/2

LOCATION	B	B_L
AT FLAP (NO SPOILER)	0.296A	0.068A
AT FLAP (INBD./OUTBD. OF SPOILER)	SPOILER CHORD	0.068A
AT AILERON	0.10A	0.10A
AT RUDDER/ELEVATOR	0.10A	0.10A

INBOARD OF AILERON (NO FLAP)

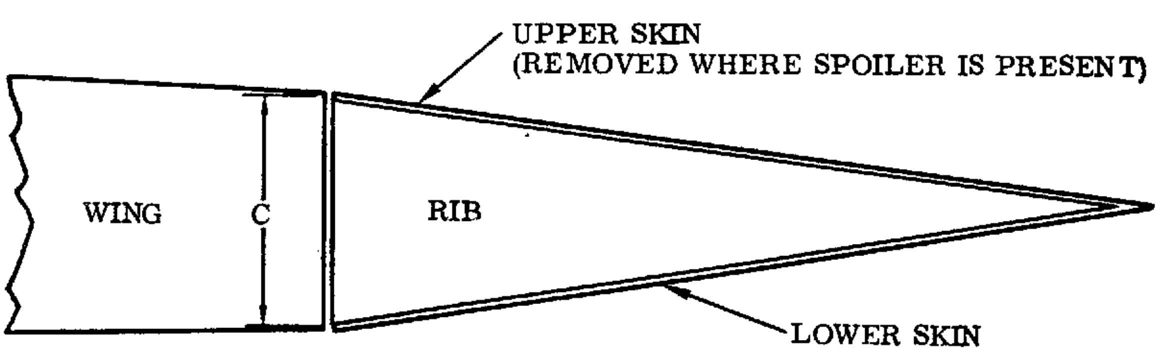


Figure 2-24. Fixed Trailing Edge

Inboard skin width, $DNSWI = K \left[\frac{2\,(DCSWI) - .28\,(DCSWI - DCSWO)}{2}\right] (.15)$

Center skin width, $DNSWC = K \left[\frac{2\,(DCSWI) - (DCSWI - DCSWO)}{2}\right] \quad (.15)$

Outboard skin width, $DNSWO = K \left[\frac{DCSWO + DCSWI - .72\,(DCSWI - DCSWO)}{2}\right] (.15)$

where

K = 2.98 for slats

= 2.57 for other surfaces

DCSWI = inboard chord length of surface

DCSWO = outboard chord length of surface

Computation of the front spar and hinge rib cap material weight assumes an additional 5.1 cm (2.0 in.) on the extrusion length. Rear spar material weight assumes an additional 1.27 cm (0.5 in.) on all sides of the flap pattern dimensions. Material weight for the skins is computed as the actual weight plus 1.27 cm (0.5 in.) of additional material on all edges. Of the total skin rivets 32% are assumed to be in each of the inboard and outboard skins, and 36% in the center skin.

Airload ribs are bent-up sheet metal and material weight is based on the flat pattern dimensions plus an additional 2.5 cm (1.0 in.) in both length and width. Theoretical and actual rib weights assume lightening holes with diameter equal to 75% of average rib height spaced a 1-1/2 diameters.

The nose ribs are assumed to be parabolic. Material weight is based on 2.5 cm (1.0 in.) added to the length and width of the flat pattern dimensions. Each rib contains one lightening hole with a diameter equal to 75% of the smaller rib chord length or 84.5% of rib height. The hinge rib webs are a solid web with no lightening holes. Material weight is calculated assuming 1.27 cm (0.5 in.) of additional material on all edges. The honeycomb trailing edge wedge theoretical weight is computed as the theoretical weight times the honeycomb core factor from the stress analysis routines. Material weight is computed assuming a honeycomb block with dimensions equalling the largest web dimensions plus 2.5 cm (1.0 in.) and adhesive weight.

The parts definition process for the tip assumes the geometry and part dimensions shown in Figure 2-25. Actual weight for the skin is computed from:

$$WT = 30\ (0.032)\ (TIP\ CHORD)\ (DENSITY)$$

The material weight for all sheet metal parts assumes an additional 2.5 cm (1.0 in.) of material on both the length and the width. All attachments assume a single row of 0.48-cm (3/16-in.) diameter rivets spaced at four diameters.

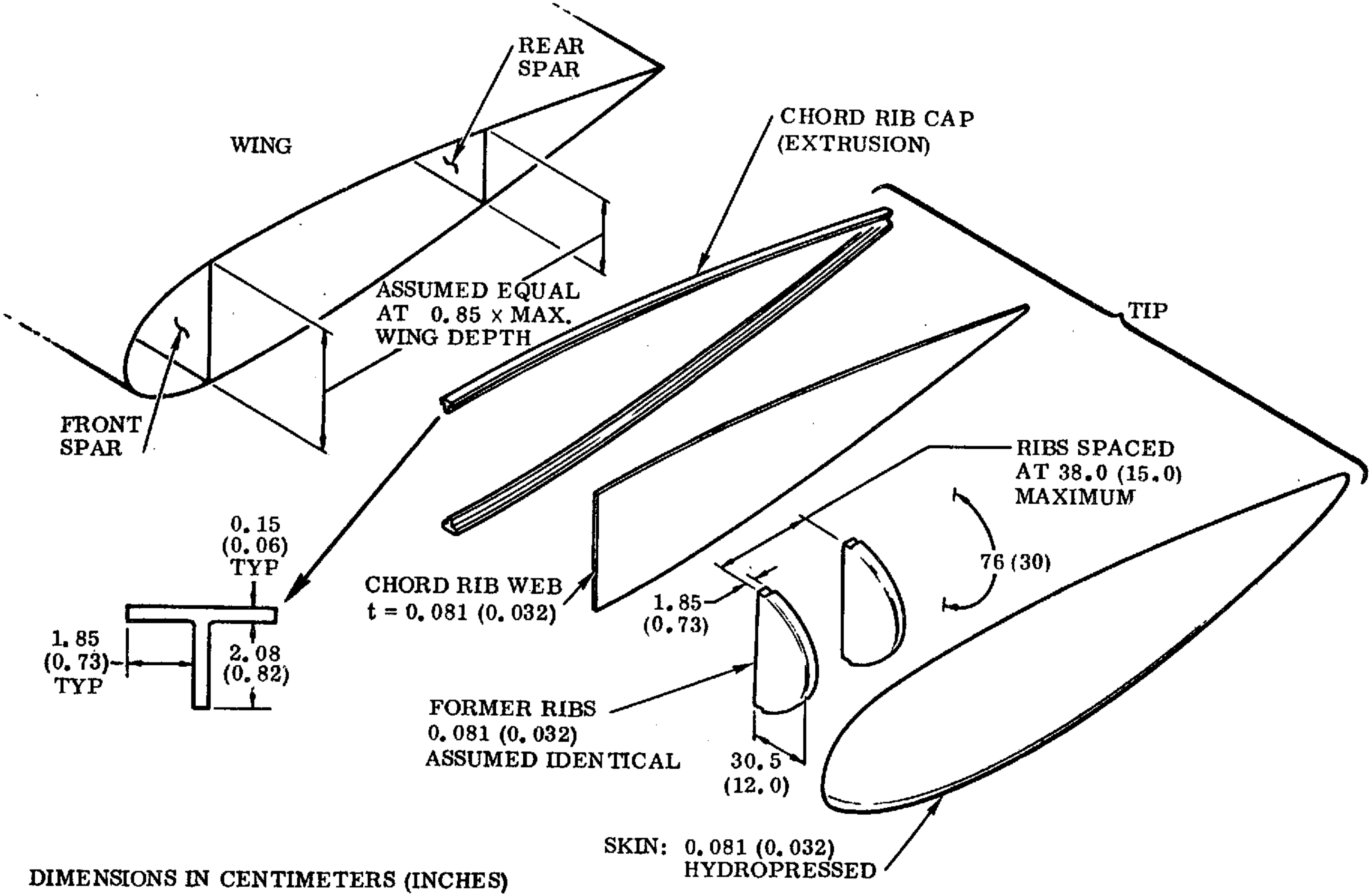


Figure 2-25. Wing Tip

2.2.2 FUSELAGE. The structural synthesis and part definition analysis associated with the fuselage is comprised of two primary operations. The first of these is a treatment of the basic fuselage shell structure, which is synthesized in terms of the required fuselage barrel sections, and component skin panels and frames, and sub-component detail parts. The second operation is a treatment of the fuselage penalty items including windows, doors (passenger, cargo, and landing gear), floors, etc. The general approach to the synthesis of the basic shell is similar in methodology to that for the aerodynamic surface box structure. Fuselage penalty items are accounted for by utilizing a statistical basis for weight, and a unit cost basis for manufacturing cost. A discussion of the actual fuselage synthesis and detail part analysis is presented in the following sections.

2.2.2.1 Fuselage Shell Analysis. Synthesis of the major fuselage shell components is carried out by the multi-station analysis subprogram, APAS, which is fully documented in Reference 10. The approach to the synthesis process is essentially the same as that discussed earlier for the airfoil surface synthesis subprogram BOXSIZ. The fuselage shell structure is assumed to have a reasonable degree of continuity and a well-defined elastic axis. Descriptive routines provide an accurate geometry and internal loads representation. Optimization of the structural elements to provide a fully stressed design is accomplished by the use of a combination of analytical and nonlinear programming techniques. Several failure modes and physical design constraints may be recognized. Output includes internal loads data, general fuselage geometry data, and member sizes and (theoretical) weights.

The basic philosophy behind a multi-station analysis is that a set of structural elements can be derived that will satisfy given design criteria at each station. It is assumed that an aggregation of these elements will result in a reasonable representation of the structure. The primary design criterion is that the structure support the applied external loads. Other common criteria are the use of a particular material or mode of construction, and minimization of structural weight. An implicit assumption of the analysis process is that systematic revision of the elements and redistribution of the material does not significantly alter the external loads distribution.

The fuselage synthesis process requires the specification of section geometry at several control stations to account for body contour variations. Twenty control stations are permitted. Station geometry is described by locating the coordinates of nodal points on the section contour. Each cross section may contain up to 20 nodes and up to four torsional cells. Section geometry is illustrated in Figure 2-26.

A total of 12 fuselage element construction modes is available. These are illustrated in Figure 2-27. The user may specify any number of different elements around the fuselage section by selecting the included node points. Frame pitch may also be specified at each station. Geometric properties are computed for each non-control station frame. No modification of the basic structural description is undertaken by the synthesis or optimization procedures. Local element sizes and dimensions are the program variables.

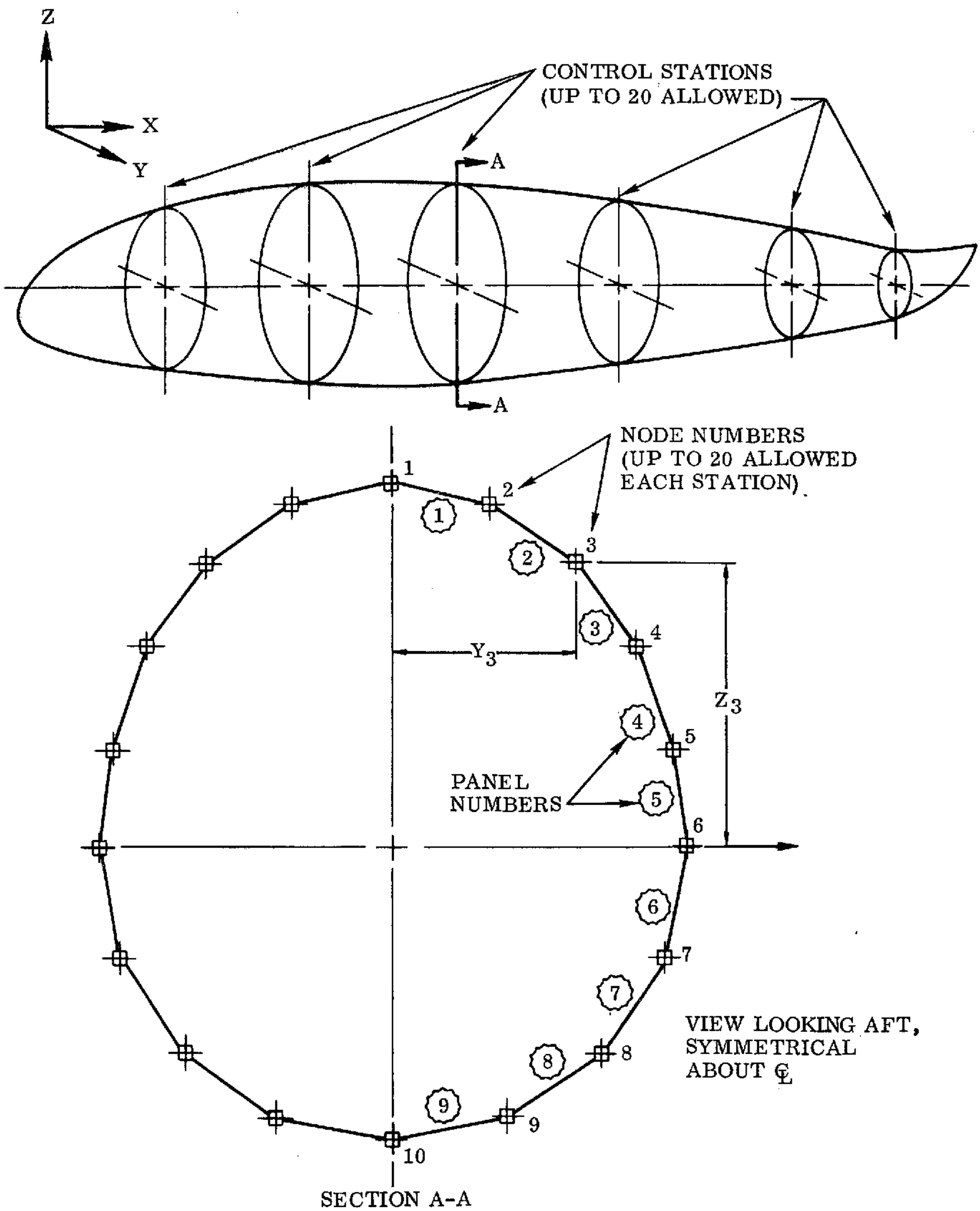


Figure 2-26. Fuselage Structural Synthesis Control Station Geometry

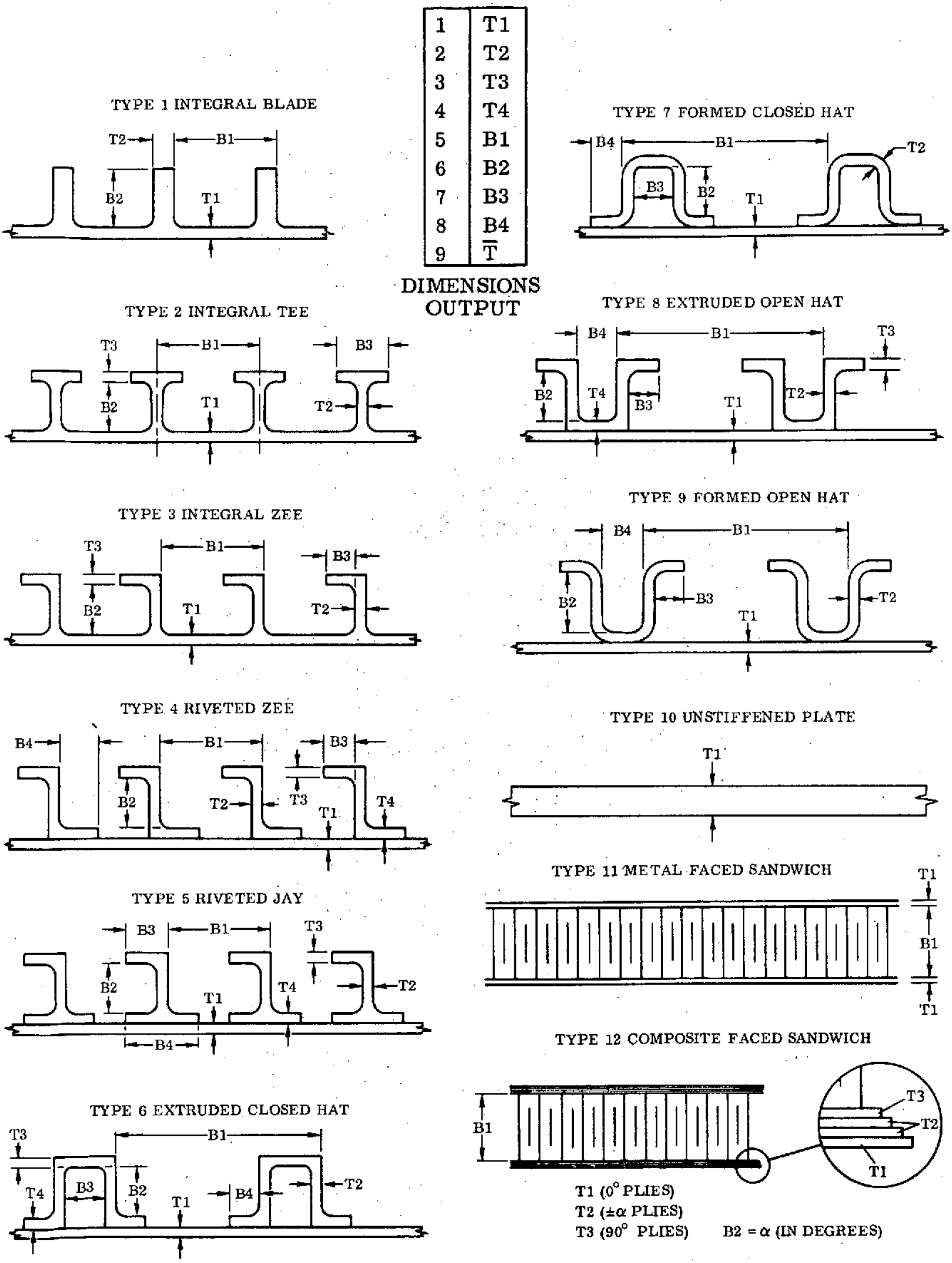


Figure 2-27. Panel Element Geometry

Material properties, including density, elastic and shear moduli, and allowable tensile, compressive, and shear strengths, are stored within the program as a function of temperature. Eleven different material types, listed in Table 2-10 are available. The program provides the capability for direct input of additional material properties as a function of temperature for both metallic and composite materials. A single material selection is allowed.

Table 2-10. Fuselage Structural Synthesis Material Selections

1. Metallic Material Input Option
2. Aluminum 2219-T87
3. Titanium Ti-8Al-1Mo-1V Duplex Annealed
4. Aluminum 2024-T6
5. Aluminum 7075-T6 Extrusion
6. Inconel 718 Plate
7. Inconel 625 Mill Annealed
8. Titanium Ti-6Al-4V Annealed Plate
9. Aluminum 2024-T851
10. René 41
11. Composite Material Input Option
12. Narmco 5505 Boron-Epoxy
13. Narmco 5206 Graphite-Epoxy

A flow diagram of the fuselage structural synthesis subprogram is illustrated in Figure 2-28. The routines utilize an analysis/design refinement iterative process. One station is operated on at a time, proceeding from nose to tail. Each loading condition is processed and the full complement of structural elements at that station are satisfactorily optimized before subsequent stations are considered. The program requires an initial design point for the first analysis loop. An estimate of the cross-sectional properties may be used or the variables may be set to unity. A summary of the panel elements and the necessary geometry data is shown in Figure 2-27. Each dimensional variable may also have a range specified by maximum and minimum values. The limits on this range subsequently become constraints during the optimization process. These constraints are a practical way of specifying minimum gage or constancy of other features such as stringer pitch. Fully effective material is used to compute section properties.

External Loads. A loading condition consists of a set of three forces and three moments (Px, Py, Pz, Mx, My, Mz) for each specified station along the structure. Present program capacity will accept up to six of these load conditions plus temperature at each station. Some convenient reference axis such as the fuselage centerline is usually adopted. Internal computations automatically transfer these loads to the section elastic axis. A stop in the loads distribution may be simulated by applying two sets of loads at the same station.

Internal Loads. Internal distribution of loads is calculated by a multi-cell box beam analysis. Complex bending stresses are found under the common assumption that plane areas remain plane (Mc/I). Torsional moment is assumed to have a T/2A distribution and direct shear is presumed to follow a VQ/I distribution. The internal member stresses, resulting from the internal loads, are used in element margin of safety calculations. Analysis routines are used to find the allowable stresses, the other necessary values in the margin of safety calculations. These routines are provided for

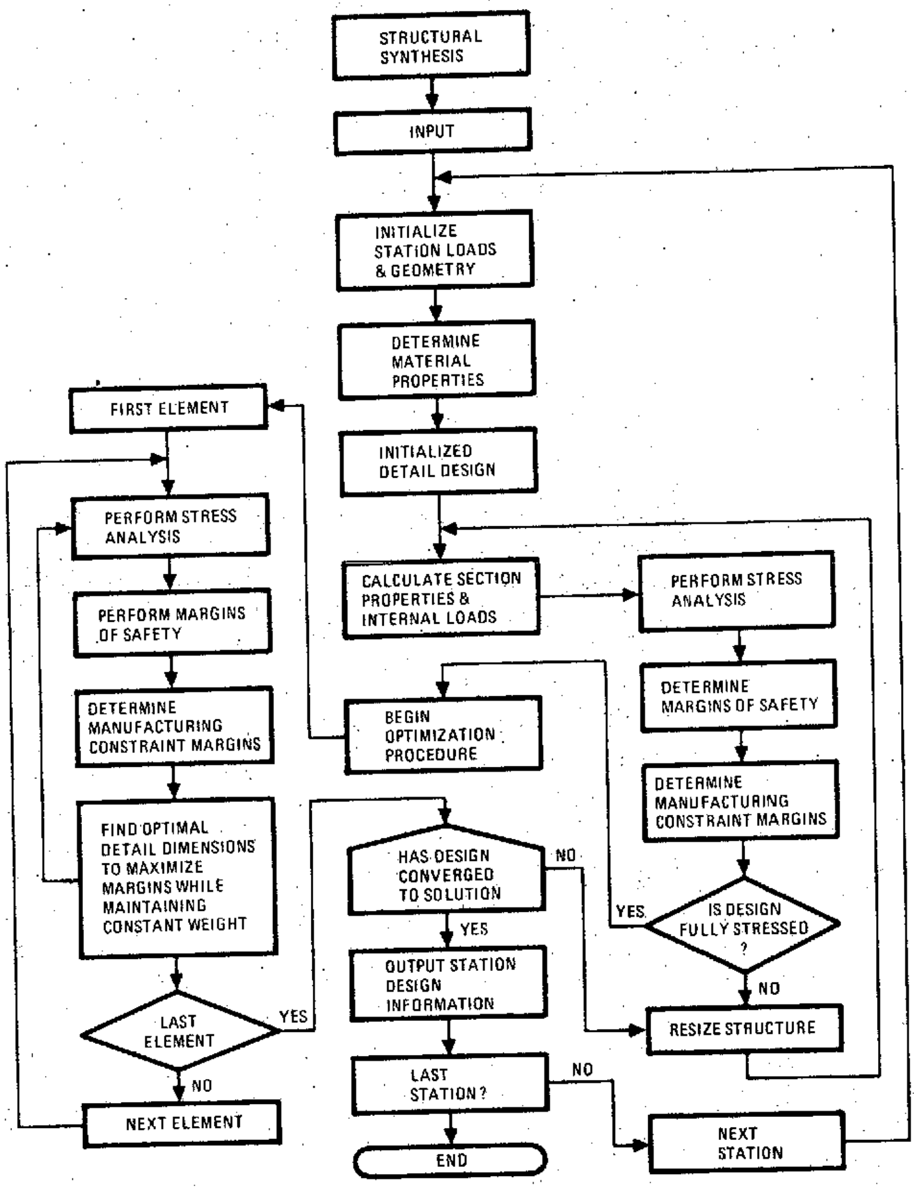


Figure 2-28. Flow Diagram for the Fuselage Structural Synthesis Subprogram APAS

several different kinds of elements and reflect the failure modes to which the components are susceptible. Buckling, crippling, and net tension are typical failure modes.

The optimization procedure is a two-step process. In the first step margins of safety are found (as previously described) for the initial size estimate. Thickness variables are adjusted up or down until each element at the particular station has a zero margin of safety. The second step of the process uses nonlinear programming techniques to maximize the margins of safety of each element. In this phase all unconstrained variables are systematically altered to produce the greatest possible positive margin. The gross element cross-sectional area, and consequently the applied stress, is held constant for this operation. At the conclusion of the second step a new internal distribution of loads is computed and the optimization process repeated. Step one alters the volume of material and step two redistributes the material in as efficient a manner as possible.

When the optimization criteria have been satisfied (e.g., a 2% or less volume change), analysis advances to the next frame station. Initial size inputs at the second and subsequent stations are the optimized results of the preceding station. Since the nonlinear optimization phase is operating on one element at a time, the number of unconstrained variables is small. Present program capacity permits a maximum of eight descriptors for each element. This may be further reduced by inputting explicit constraints. As a result solution convergence is rapidly achieved.

Fuselage load conditions in the yaw plane are almost always fully reversible. The result is a structure symmetrical about a vertical centerline. This structural symmetry may be preserved, without running extra load conditions, in the synthesis program by a special symmetry grouping feature. A priori definition of symmetry groups is required. Minor image elements are usually, but not necessarily, collected in these sets. This group is sensitive to failure modes resulting from all loads on any of the elements in that group. Final sizes will reflect the optimum element that may be used in any of the symmetry group positions.

2.2.2.2 Fuselage Shell Part Definition. The structural synthesis routines produce general fuselage geometry at each control station. Data generated for each station includes barrel perimeter, frame spacing, panel cross-section dimensions, panel stiffener spacing, etc. The parts definition routines take the output from the fuselage structural synthesis and derive the detail parts sufficient to construct the complete basic fuselage shell structure.

The first step in the parts definition process is to develop the geometry data to a greater degree of detail. Data output from the structural synthesis for various control stations are interpolated to provide data for actual fuselage stations. The following geometry data are derived:

a. Fuselage frame spacing and frame stations.

b. Fuselage barrel lengths and number of barrels.

c. Barrel perimeters.

d. Complete frame cross-section geometry and perimeters.

e. Frame segment length and number of segments.

f. Panel width and number of panels around circumference.

g. Complete panel cross-section geometry.

h. Window cutout dimensions.

The structural synthesis derives frame spacing at given control stations. This spacing may vary from station to station. The parts definition places a frame at the first control station, and frames between the midpoints of adjacent control stations are given the spacing of the nearest control station. A frame is placed at the first and last control station (aircraft nose and tail) only if its perimeter is not zero. Perimeters for each frame station are computed by interpolating between control station perimeters. The number of frame segments may be input or computed. In the absence of an input two segments are assumed for a maximum control station barrel perimeter of less than 1143 cm (450 in.). Otherwise three segments are used.

The fuselage barrel length is initially determined by either user input (barrel length or number of barrels) or by dividing the fuselage into equal barrel lengths. A maximum length of 1006 cm (33 ft) is allowed. The nose and tail barrels are half the length of the others. Barrel lengths are then adjusted to fall halfway between frame stations. Barrel perimeters are computed by interpolating control station perimeters from the structural synthesis.

The number of panels or panel length ratios may be input by the user. For computation of skin panel width the fuselage cross section is broken up at nodes into individual panels. If the entire cross section is one "symmetry group" (all the same construction and subject to the same structural synthesis geometry constraints), it is broken up into an even number of panels with a width at the largest control station perimeter of 226 cm (89.1 in.). If more than one "symmetry group" occurs at a given cross section, panel widths are defined such that only one "symmetry group" is contained in a given panel. All cross sections are assumed to have the same number of panels (minimum of 4 and maximum of 10), and all panel end points are on the same node. Panel lengths are assumed to be equal to the barrel lengths except for the nose and tail barrels where the effect of fuselage taper is accounted for. The end width of each panel is computed, and a mid-height panel on each side of the fuselage is designated for containing windows. Panel end cross-section dimensions (Figure 2-27) and average cross-section dimensions are computed by interplating between control stations.

The actual parts definition process is comprised of four steps. First, the complete skin panel assembly is derived: the corresponding parts are skin, stringers, and ripstops. Second, the complete frame assembly is derived, comprised of frame segments,

frame splice angles, shear clips, and shear clip splice plates. Third, the parts necessary to assemble each fuselage barrel section are derived, including internal and external longitudinal skin panel splices, intercostals, and intercostal clips. And fourth, the parts required to assemble the barrel sections into a complete fuselage shell are derived: stringer splices, barrel finger splices, barrel strap splices, and splice plates. For each detail part originated a theoretical weight (OPWT), an actual weight (ACWT), and a raw material purchase weight (MAWT) is computed. Fasteners are accounted for as each group of parts is brought together to form an assembly.

A typical skin panel assembly is illustrated in Figure 2-29. The structural synthesis routines optimize the shell structure at individual control stations. This normally produces different quantities of stringers (or risers) at each station. Transport aircraft always have a constant number of fuselage stringers because of the difficulty of transferring discontinued stringer loads to adjacent stringers. Therefore, a constant number of stringers is assumed for a given panel at any station location. For each panel a maximum number of stringers is determined by dividing the panel width by the stringer spacing at each end of each barrel. This number is used for that panel at all fuselage station locations.

Windows are assumed located in the specified mid-height panels between each frame for all but the nose and tail barrel sections. Window cutout dimensions are computed as follows. Width is assumed equal to 60% of the local frame spacing with a maximum width of 64 cm (25 in.) specified. Height is assumed equal to 1.35 times the window width. The arrangement of a typical window is presented in Figure 2-30.

The theoretical and actual weights for the integrally stiffened skin panel are computed using an average panel length, average panel width, and equivalent flat plate thickness averaged for each end of the panel. The material weight can be expressed in terms of the maximum cross sectional area of the largest end of the panel with an additional 0.25 cm (0.10 in.) of material added on all sides of the cross section to account for machining.

For skin-stringer skin panels the skins and stringers are considered separately. The theoretical skin weight is based on the average of the tapered skin thicknesses. Actual weight is based on a standard sheet gage shown in Table 2-11, which is equal to or larger than the maximum thickness of a given panel. Both theoretical and actual weights account for window cutouts. Material weight assumes a standard sheet gage, and average panel lengths and widths with an additional 5.1 cm (2.0 in.) of material along all the edges.

Table 2-11. Standard Sheet Gages

cm	in.	cm	in.
0.081	(0.032)	0.203	(0.080)
0.091	(0.036)	0.229	(0.090)
0.102	(0.040)	0.254	(0.100)
0.114	(0.045)	0.318	(0.125)
0.127	(0.050)	0.406	(0.160)
0.160	(0.063)	0.483	(0.190)
0.180	(0.071)	0.635	(0.250)

Theoretical stringer weight assumes a tapered stringer. Actual weight assumes a constant section stringer with the dimensions of the

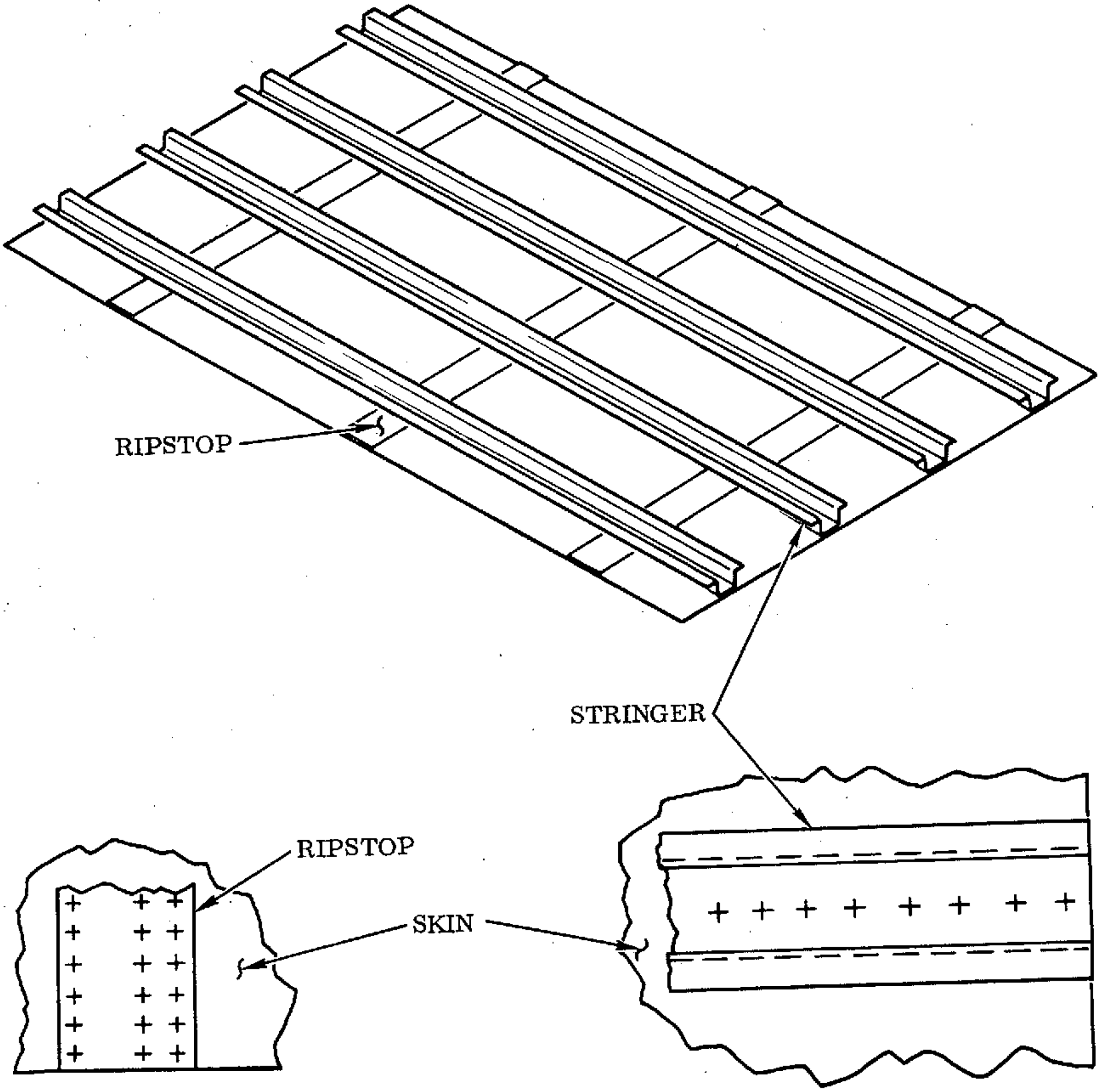
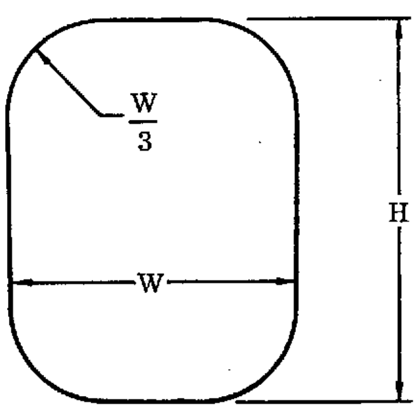


Figure 2-29. Skin Panel Assembly



W = 0.60 * FRAME SPACING
H = 1.35 * W
$$\text{AREA} = H * W - \frac{W^2}{9}(4-\pi)$$

Figure 2-30. Typical Window Arrangement

end with the largest cross-sectional area. The material weight for extruded stringers utilizes the same cross-sectional area as the actual weight computation, but assumes an additional 5.1 cm (2.0 in.) of length for cutoff. For sheet metal stringers, the actual weight is based on a standard sheet gage equal to or larger than the maximum stringer thickness. The material weight is calculated in the same manner as the actual weight with an additional 2.5 cm (1.0 in.) of flat stock on the width and 5.1 cm (2.0 in.) on the length.

Thick plate skin panels and sandwich panel face sheets are assumed to be tapered sheet or plate. Theoretical and actual weights are based on the tapered material thicknesses. Material weight assumes the maximum thickness and an additional 5.1 cm (2.0 in.) added to the panel length and width. The honeycomb core for the sandwich panel is treated in a like manner.

Table 2-12. Rivet Sizes

Component Thickness		Rivet Diameter	
cm	in.	cm	in.
		0.318	(0.125)
0.091	(0.036)		
		0.396	(0.156)
0.114	(0.045)		
		0.478	(0.188)
0.318	(0.125)		
		0.635	(0.250)
0.381	(0.150)		
		0.792	(0.312)
0.508	(0.200)		
		0.953	(0.375)

Skin panel assembly assumes rivet sizes based on the skin thickness, as shown in Table 2-12. For integrally stiffened panels the panel average skin thickness is used to choose rivet diameter. For skin-stringer constructions the standard sheet gage is used. For plate and sandwich constructions the maximum total skin thickness is used.

Ripstops are thin sheet metal doubler straps (often made of titanium) that lie on the skin under each frame. Their purpose is to stop fuselage skin fatigue cracks from growing. They are assumed to have the same thickness as the skin and a length equal to the panel width. Ripstop width is determined by fastener spacing and edge distance requirements. Ripstop-to-skin rivets consist of three rows, as shown in Figure 2-29, spaced at four diameters. The fourth row is supplied by the frame shear clip-to-skin rivets called out in the frame parts definition analysis. Stringer-to-skin rivets are placed in one or two rows as depicted in Figure 2-29. Rivets are spaced at four diameters pitch.

A typical frame assembly is illustrated in Figure 2-31. The frame cross-sectional area is computed using loads and materials property data from the structural synthesis routines. An expression for the frame cross-sectional area is as follows (Reference 11).

$$\left[\frac{C_f * D^2 * M}{K_4 * E_F * L}\right]^{1/2}$$

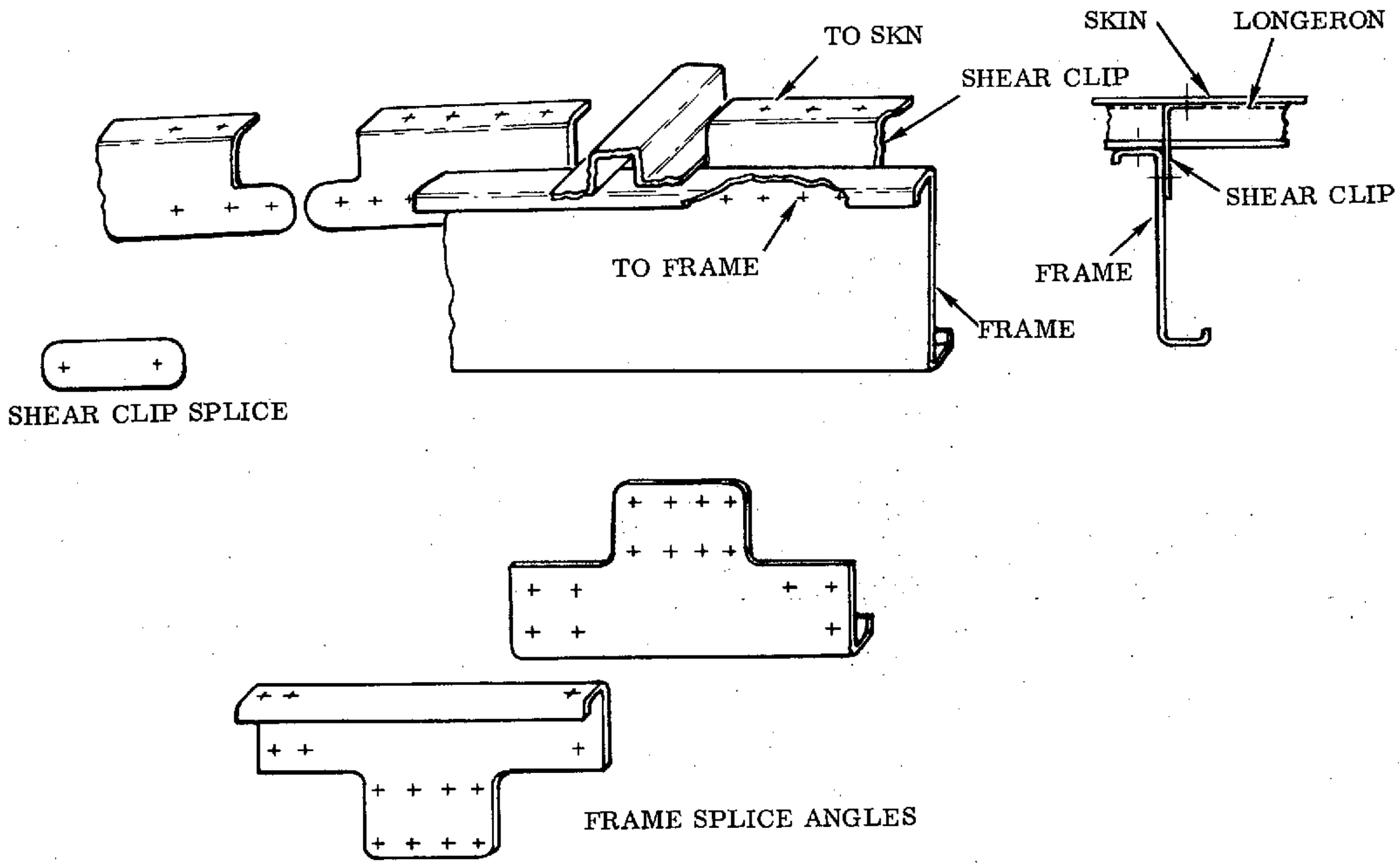


Figure 2-31. Frame Parts

where

D = Shell diameter (use body width)

M = Fuselage bending moment (use maximum value from structural synthesis)

L = Frame spacing (from parts definition geometry computations)

C_f = Coefficient (use 0.0000625 from Shanley)

K_4 = Shape parameter (use 5.4 from Shanley)

E_F = Frame modulus of elasticity (from structural synthesis)

The computed frame area is used to derive the frame dimensions by an iterative technique. Frame height and flange width dimensions are sized based on fastener spacing and edge distance requirements (Figure 2-32) with a minimum frame thickness of 0.102 cm (0.040 in.) specified. The computed (or minimum) frame thickness is used to determine theoretical weight, while a standard gage equal to or greater than the computed frame thickness is used to determine actual weight. The material weight computation assumes a standard gage with 5.1 cm (2.0 in.) of additional width and a length equal to the frame segment length plus 10.2 cm (4.0 in.).

It is assumed that there are two shear clips per frame segment, which attach the frames to the skin. The long shear clip is equal to the frame segment length minus the length of two stringer spacings. The short shear clip spans the frame splices, and is two stringer spacings long. Shear clip cutouts for stringers are derived on the basis of the largest computed value for stringer height and width. Thickness is assumed to be equal to the frame thickness. Theoretical and actual weights are computed in the same manner as for frames; material weight is computed assuming a standard gage and 2.5 cm (1.0 in.) of additional width and 5.1 cm (2.0 in.) of additional length.

Shear clips are spliced together with a shear clip splice plate (Figure 2-31) that is assumed to have the same thickness as the shear clip. The length and width are sized for picking up a single row of four rivets. Two frame splice angles (Figure 2-31) are assumed for each frame segment splice. These angles nest inside the frame at the splice and are assumed to be the same thickness as the frame. The length of the angles is made equal to a stringer spacing plus a stringer width.

Frame thickness is used to size all the fasteners required in the frame assembly (Table 2-13). One row of fasteners is assumed through the shear clips into the frames, and two fasteners are assumed to attach each stringer to a frame. A single row of fasteners from the skin through the shear clip is assumed. Typical fastener spacing has been defined as four diameters.

The detail parts required to splice the skin panel and frame assemblies into a barrel section include internal and external longitudinal panel splices, frame stabilizing intercostals, and intercostal clips. The assembly is illustrated in Figure 2-32.

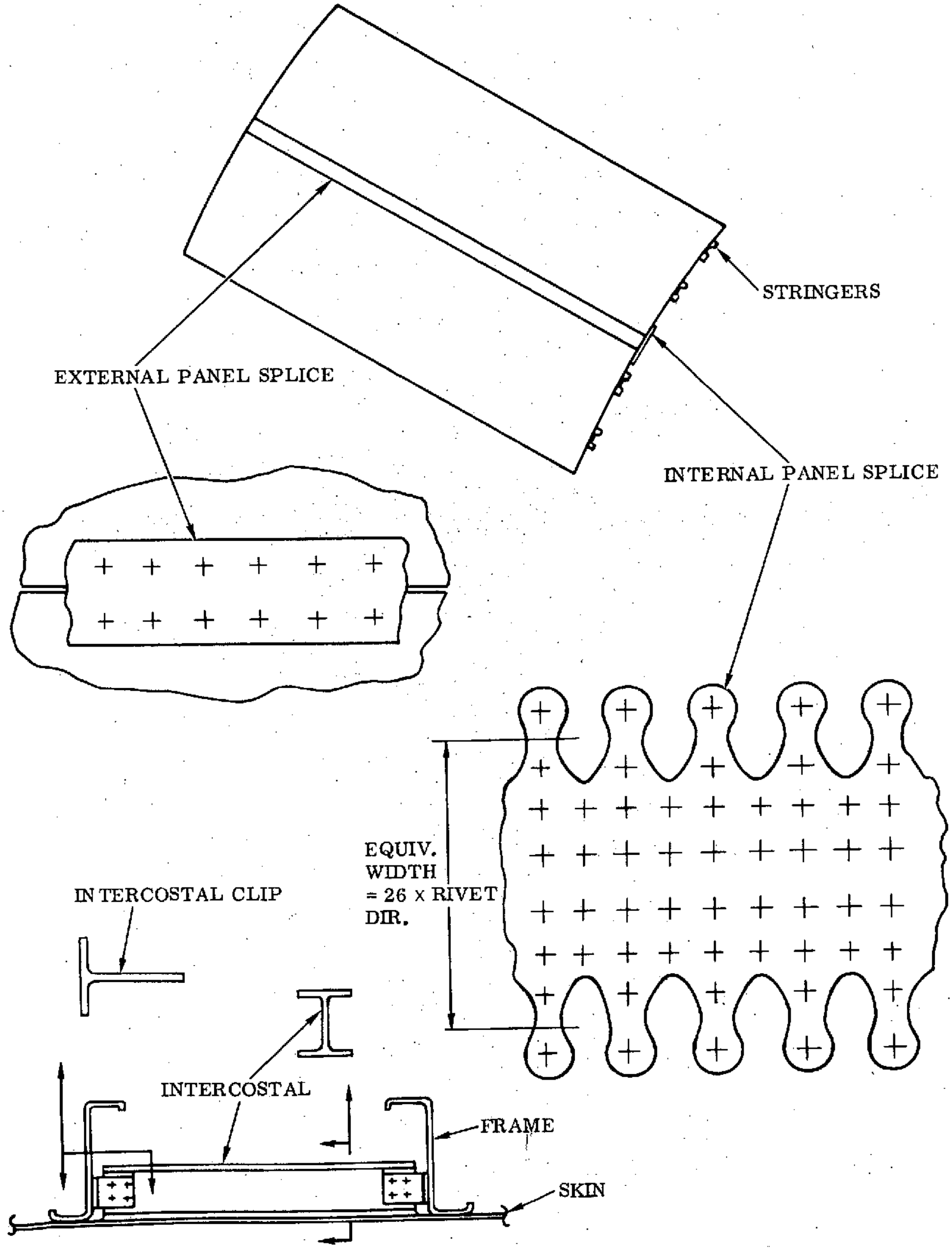


Figure 2-32. Barrel Parts

The external panel splice is a flat plate splice running the length of the skin panel (which is equal to the length of the barrel). The width is 10 fastener diameters and the thickness is set equal to the skin thickness. Theoretical and actual weights are assumed to be equal. Material weight is computed assuming 1.3 cm (0.5 in.) of additional width and 10.2 cm (4.0 in.) of additional length.

The internal panel splice is scalloped with fingers, but is synthesized as a straight-edged plate with an equivalent width of 26 skin fastener diameters. The actual weight is assumed equal to the theoretical weight, which is based on a standard sheet gage equaling or exceeding 40% of the skin thickness. Material weight assumes 10.2 cm (4.0 in.) of additional length and 5.1 cm (2.0 in.) of additional width. The internal splice is assumed to be attached to the skin with the equivalent of four rows of fasteners spaced at four diameters. The two middle rows also pick up the external splice on the skin exterior.

Intercostals are extruded I-sections located in every other frame bay between stringers and spaced five stringers apart. Their length is equal to the frame spacing minus a clearance of 1.0 cm (0.4 in.). Height of the intercostal section is assumed to be 40% of the intercostal height. Theoretical and actual weights are based on a thickness equal to the maximum fuselage skin thickness. Material weight is computed assuming the same cross-sectional area and an additional 5.1 cm (2.0 in.) of length. Intercostal to skin fasteners are assumed to be comprised of two rows of skin fasteners spaced at four diameters.

Two extruded tee clips attach each intercostal to frames. The length of each extruded clip is such that it fits between the intercostal flanges (Figure 2-32) with a total clearance of 0.38 cm (0.15 in.). The height of the clip measured from the frame toward the intercostal is set equal to the length. The flange against the frame has a width equal to eight fastener diameters. Thickness is assumed equal to the intercostal thickness for all weight computations. Intercostal clips are attached using frame fasteners, six through the frame and four through the intercostal.

The detail parts required to splice the barrel sections into a complete fuselage section include stringer splices, barrel finger splices, barrel strap splices, and splice plates. The assembly is illustrated in Figure 2-33. The stringer splice cross sections are shown in Figure 2-34.

2.2.2.3 Fuselage Penalty Analysis. The treatment of fuselage penalty items encompasses windows, doors (landing gear and side loading), and floors. The analysis is comprised of a statically based actual weight computation and a unitized manufacturing cost computation. The values derived for fuselage penalty weights and costs are added to those of the basic fuselage shell (which are determined from a structural synthesis/parts definition analysis) to obtain total fuselage data.

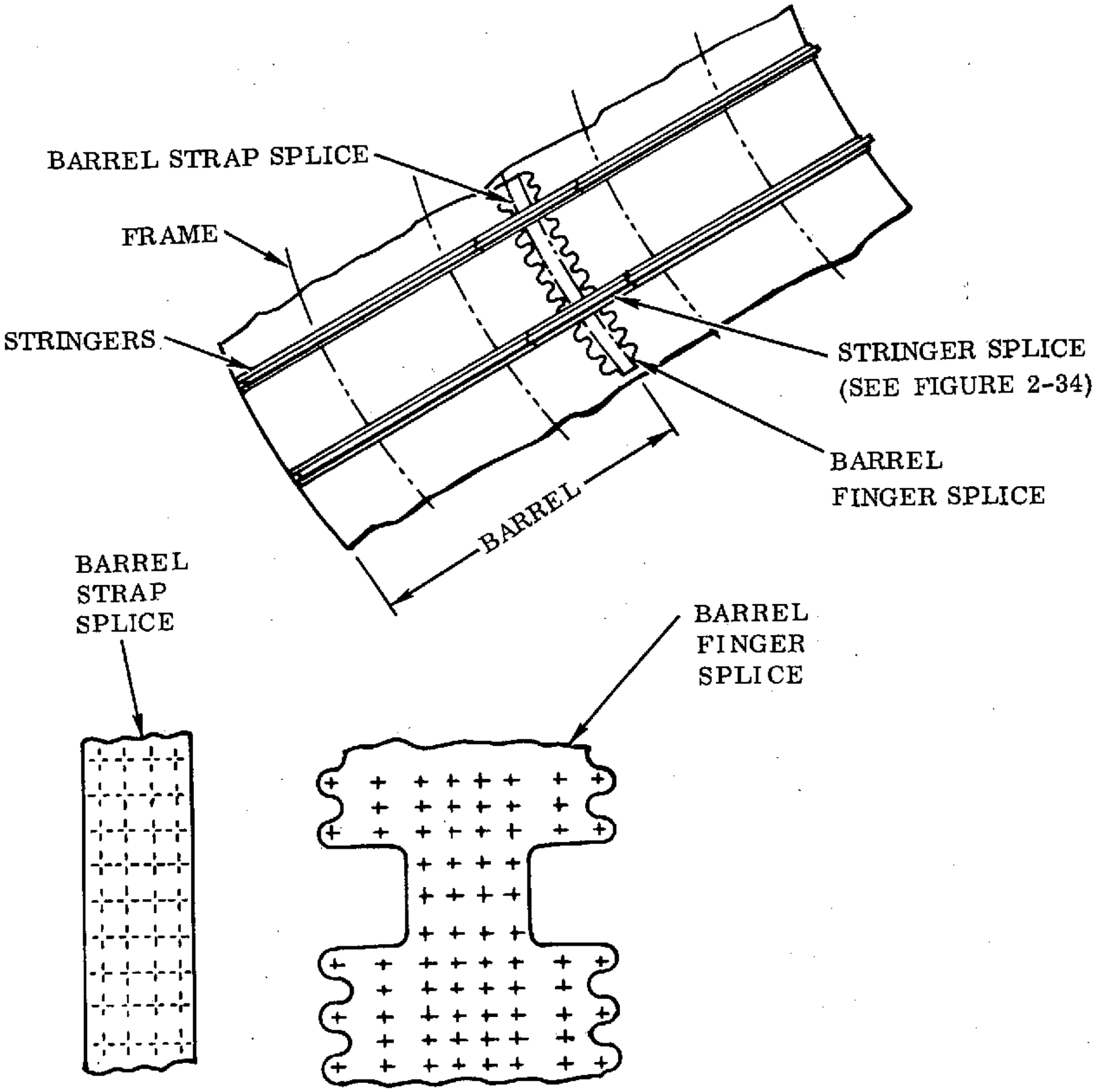


Figure 2-33. Barrel Splice Parts

Window weights are computed as a function of the total glass area required for the specified number of windows. Individual window area is computed from the assumed window geometry illustrated in Figure 2-30. Windows are assumed located between each frame for all but the nose and tail barrel sections, and hence, the number of windows is equal to twice the number of frames in those barrel sections minus two. Following is the equation used to compute the total window weight penalty (Reference 12):

WNDWWT = 10.0 * AGL

where

WNDWWT = Window weight and AGL = Total glass area

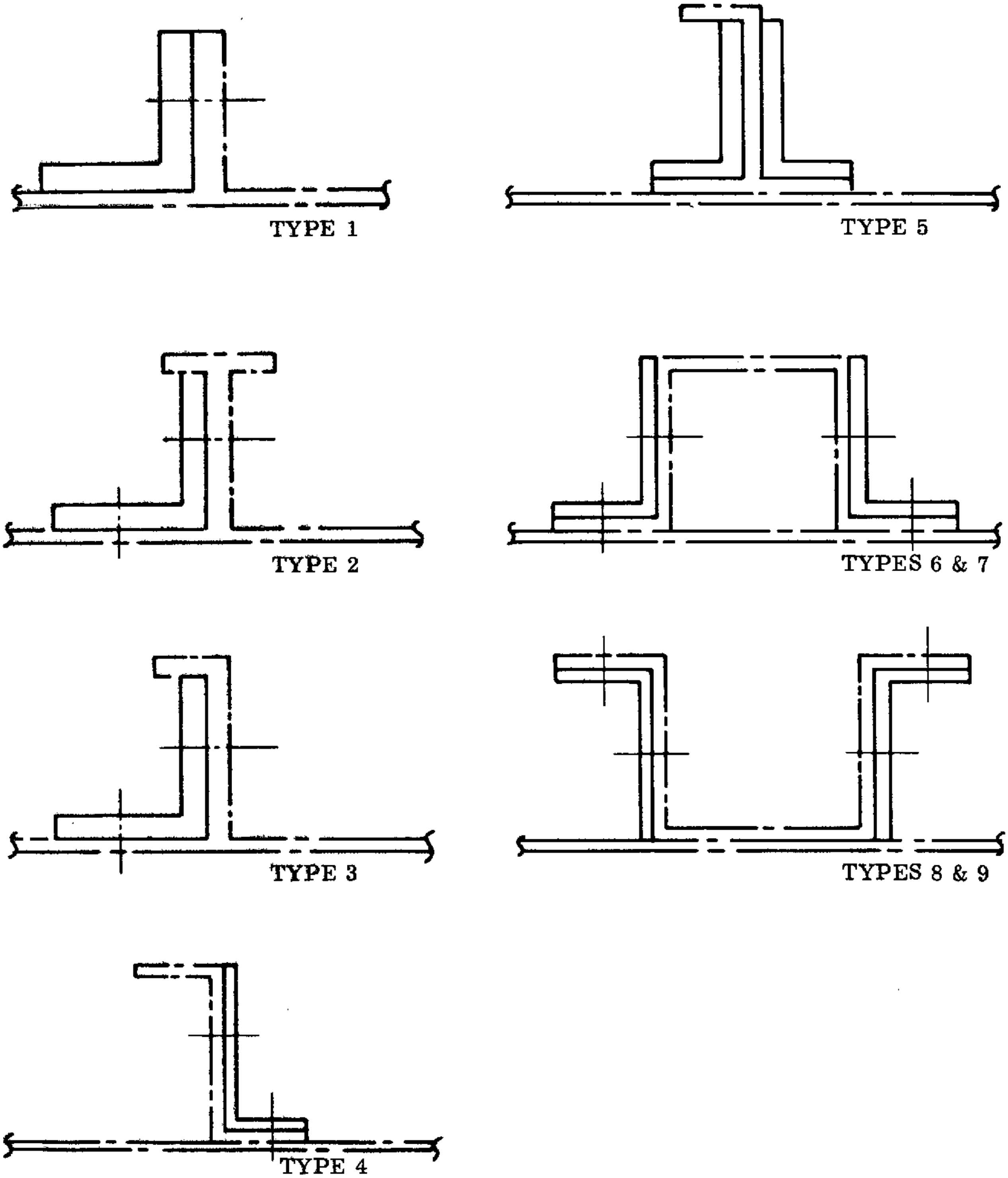


Figure 2-34. Stringer Splice Cross-Sections

Doors are assumed to include nose and main landing gear doors, and side loading cargo and passenger doors. Nose landing gear door weight is computed as a function of the maximum dynamic pressure and the total door area. Main landing gear weight and side loading door weight are computed as a function of the total door area alone. The following equations are used for the weight computations (Reference 12).

NLGWT = .44 * QMAX ** .3 * SND

MLGWT = 3.223 * SMD ** 1.125

SLDWT = 9.0 * SDA

where

NLGWT	=	Nose landing gear door weight
QMAX	=	Maximum dynamic pressure
SND	=	Nose landing gear door area
MLGWT	=	Main landing gear door weight
SMD	=	Main landing gear door area
SLDWT	=	Side loading door weight
SDA	=	Side loading door area

The weight of floors and floor supports is computed as a function of the ultimate flight design load factor, a design floor loading, and the floor area. The equation is as follows (Reference 12).

FLRWT = 6.51 * (NZ * WF * AF/1000) ** .924

where

FLRWT	=	Floor and floor support weight
NZ	=	Ultimate flight design load factor
WF	=	Design floor loading at 1.0 g
AF	=	Floor area

The floor and window areas can be computed from the fuselage shell geometry. Values for the maximum dynamic pressure and the ultimate load factor are brought across from the vehicle synthesis portion of the program. The user may input values for the design floor loading, and the nose gear, main gear, and side loading door areas. In the absence of an input, typical values for a passenger transport type of aircraft are utilized. The values are: design floor loading, 3591 N/m^2 (75 lb/ft^2); nose gear door area, 1.4 m^2 (15 ft^2); main gear door area, 7.4 m^2 (80 ft^2), and side loading door area, 139 m^2 (1500 ft^2).

The manufacturing cost portion of the analysis is based on an average unit cost. The user may input a value, or in the absence of an input a value of \$176/kg (\$80/lb) is assumed. The cost is derived by multiplying the weight previously computed by the appropriate average unit cost.

2.3 COST SYNTHESIS

The cost analysis portion of the program incompasses the following: manufacturing cost, material cost, engineering cost, tooling cost, total vehicle program cost, and return-on-investment. Manufacturing cost is determined as a function of the actual shop processes necessary to produce each part. From this the corresponding number of labor hours that are required can be computed, and hence, the manufacturing cost.

Material cost is derived on the basis of the amount of raw material stock purchased, material type and form, and various extra cost items such as special lengths, widths, and tolerances.

Engineering cost is derived on the basis of equations originally developed by Levenson and Barro. Both initial and sustaining engineering costs are represented.

Tooling cost is derived on the basis of the number of dissimilar parts to be produced, and hence, the total number of tools required. Basic tooling, rate tooling, and sustaining tooling costs are represented.

Total vehicle program costs are derived using primarily the cost estimating relationships developed by Kenyon. A learning curve approach is applied to adjust first unit costs to those of subsequent units.

A return-on-investment analysis utilizes computed aircraft performance parameters and the 1967 Air Transport Association formula to derive direct operating costs. Indirect operating costs and return-on-investment are derived on the basis of an input traffic route structure. All cost data are computed relative to a specified dollar reference year. Actual cost estimation methodology is discussed in detail in the following sections. All of the required computer program input in the area of cost analysis utilizes the British system of units.

2.3.1 MANUFACTURING COST, PROCESSES, STANDARD HOURS, AND RATES

2.3.1.1 Manufacturing Cost. The technique being used to estimate first unit manufacturing costs basically is as follows. A breakdown of major vehicle components into their detail parts is accomplished through the use of vehicle synthesis, structural synthesis, and part definition operations. The actual manufacturing cost analysis is based on calculating the material, and direct and indirect labor costs associated with the fabrication and assembly of each detail part. For each part, in turn, a record of manufacturing and assembly operations required to produce that part and integrate it into the vehicle structure is established. For each operation specified the number of direct or actual labor hours required to perform the operation is derived, and based on this, direct labor and indirect overhead costs are calculated. From the part geometry, the material required for each part is derived, and based on this, material costs are calculated. Figure 2-35 summarizes the steps necessary in determining the manufacturing cost from the detail part level.

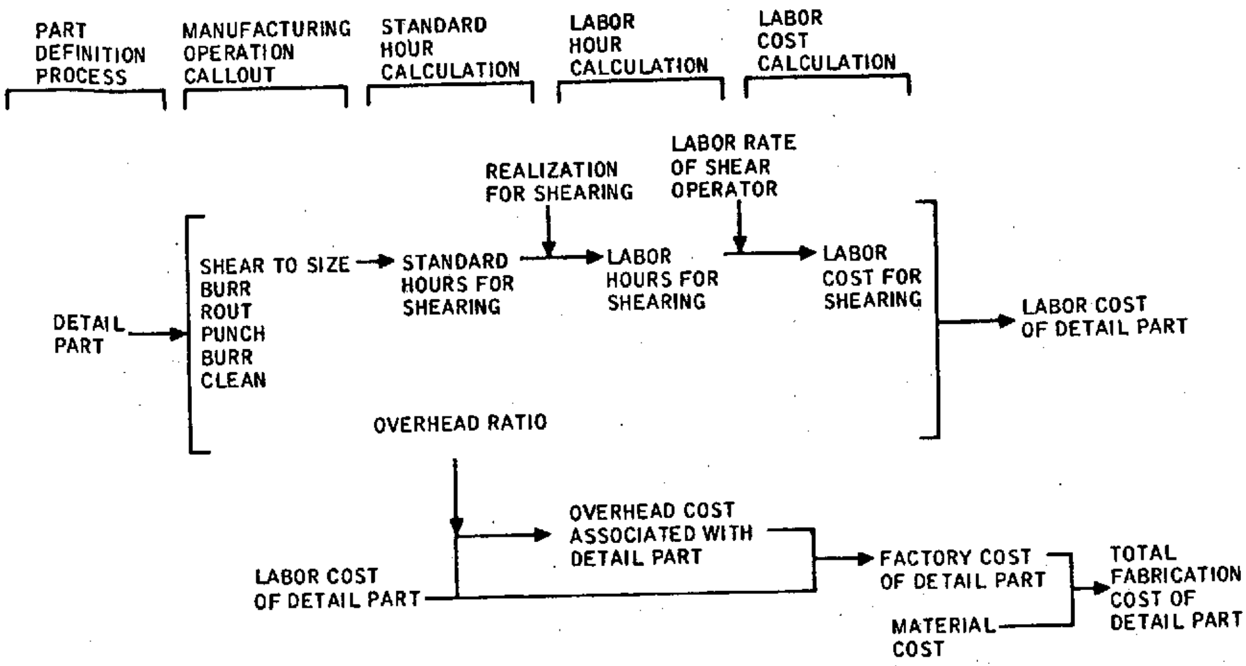


Figure 2-35. Cost Analysis Sequence Based at the Detail Part Level

The derivation of direct and indirect costs associated with the manufacture of each detail part involves the determination of the number of actual labor hours required for each production process, and the corresponding labor rates. The computation of actual labor hours is accomplished by multiplying a computed number of shop standard hours (discussed in detail in a following section) by a shop efficiency (the so-called realization) factor. Labor costs, then, are simply the actual labor hours multiplied by a representative labor rate. Overhead costs are computed by multiplying the direct labor costs by an overhead ratio that is derived from accounting practice. Each of these computations is discussed in detail in the following sections; the equations are:

LABHR = STDHR/REFCT

LACOST = LABHR * LARATE

VCOST = VRATE * LABHR * LARATE

where

LABHR = actual labor hours

STDHR = Standard hours

REFCT = realization (efficiency) factor

LACOST = direct labor cost

LARATE = labor rate

VCOST = indirect overhead cost

VRATE = overhead ratio

2.3.1.2 Manufacturing and Assembly Processes. The parts definition routines were designed to provide an accounting of the detail parts required to produce a complete airframe. Each detail part is looked at individually and analyzed in terms of the manufacturing and assembly processes associated with it. In order for this analysis to be performed it was necessary to be able to internally account for each of the required shop processes.

To develop the process lists associated with each part, a library of shop planning records was established from existing production articles. These documents were studied and used to identify the typical processes associated with each part. A method was developed to internally relate each part to its corresponding list of processes. It was the intent to provide a means of internally generating the equivalent of a shop planning order. A representative example of such an order is presented in Figure 2-36. It is from this type of document that the specific planning for the production of an individual part can be implemented.

Currently, equations for a total of 33 manufacturing and assembly operations are represented within the program. It is the purpose of these equations to compute a value for the number of standard hours necessary to perform the specified tasks. While the equations for each of the operations are strictly valid only for the specified process, a reasonable number of standard hours may be obtained by applying the equations to related processes. Provision has been made in the program for the future addition of any new processes that might be needed to account for new production processes.

As each detail part, subassembly, and assembly is considered during the part definition portion of the program, a part index (KK) is assigned. This index is associated with a program block that calls, in turn, each of the applicable standard hour equations for the part. For example, a wing rib brace is given the part index KK = 26, which is used to direct the program to the operations required to manufacture the brace. The operations specified might include the following depending on the structural mode:

SAWING:	saw the raw material stock to size
BURRING:	deburr the sized brace
DRILLING:	drill the required holes
CLEANING:	clean and degrease the finished brace
SURFACE TREATMENTS:	perform required surface treatments
PAINTING:	primer and paint finished part
IDENTIFY:	mark with identifying part number
INSPECTION:	inspect finished part

A value for standard hours is computed for each of these, and the sum is the total number of standard hours that manufacturing of this particular rib brace would be expected to require.

GENERAL DYNAMICS CONVAIR M-DAY LTR-NO PRJ PART NUMBER
SHOP ORDER PLANNING 363 D 8 4T11001-125A

PLANNER	GRP	CHG CD	ENGRG D/C	P/L	INSP CL	NO	LTR-NO	PRJ	PART NUMBER
EBROLASKI	11	RC	OD			26	D	8	4T11001-125A

TITLE	MATERIAL PERCENT	GRP NUMBER	A/Z
STIFFENER	3.000	12 120239	D

MATL DESCRIPTION	GAU/DIA	WID/WALL	LENGTH	SPECIFICATION	COND	PTS	OP/END	PT-CL
SEE NOTE A			30.000	QQ-A-200/11	NOTE	01		2

DATE	STK RM	QTY	BY	SPARE	CUST	SHOP	INSP

-----OPERATIONS-----

INSP STMP	DEPT	CC	OPER NO	OP SCH	OPERATION DESCRIPTION TOOL/SPEC NO TOOL SEQ NO	TOOL SYMB	MACH CODE	SET-UP TIME	RUN TIME
			A		LS39048-1 7075T6511 EXTRU				
			B		STP63-001 STP58-208 STP57-301				
					STP53-201 DS30004 STP59-201				
	836	07	10		SAW TO LENGTH 1-1		6880	0.12	0.0031
					4T11001-125A	SP			
			15		INSPECT				
	01	03	20		ROUT		6455	0.08	0.0256
					4T11001-125A	MAST			
					4T11001-125A	SP			
	01	02	30		DRILL PILOT HOLES (74) PLCS		1552	0.25	0.1750
					4T11001-125A	DLT			
	01	12	40		BURR		2905		0.0155
	01	12	50		IDENTIFY (TAG) STP63-001		2910	0.07	0.0008
					4T11001-125A	SP			
			55		RW				
			65		INSPECT				
	02	05	70		ALK ETCH STP57-301		5845		0.0216
	02	11	80		SULF ANODIZE STP58-208		5807		0.0144
			85		INSPECT				
	02	03	90		(I) ZCP STP59-201		4815	0.10	0.0052
	02	03	100		IDENT R/S STP63-001		2910	0.07	0.0032
					4T11001-125A	SP			
			105		INSPECT				
			115		HPS1-00156 (PROTECT)				
					TOTAL			0.69	0.2644

Figure 2-36. Example of a Shop Planning Order for a Rib Brace

2.3.1.3 Standard Hours. Standard hours are, as the name implies, a standard time, measured in hours, which represents an optimum for the time required to perform a given task. They are the number of hours it would take for a normal person to do a normal job under normal conditions. They do not include allowances for fatigue, personal needs, rest breaks, machine adjustments or tool breakage, close tolerance work, etc. Thus, the standard hours are an idealized time scale against which actual time may be compared.

Standard hours are used industry-wide for estimating purposes. They are established by industrial engineering departments through the analysis of time and motion studies carried out for standard shop operations and procedures. They are used by industrial engineering departments to estimate the time required to perform production tasks, and by accounting departments to measure performance through comparison with actual times. By being able to estimate an optimum time in standard hours and the measuring a corresponding real or actual time, relative efficiency factors (or realization factors) can be established for various shop processes and tasks.

Included as a part of the shop planning order (Figure 2-36) is an estimate for the number of standard hours corresponding to each shop operation. The program, following a parallel logic, was designed with a capability to generate a number of standard hours corresponding to each of the internally generated shop processes. This is accomplished by a series of standard hour equations derived based on standards data acquired through the industrial engineering department.

An example of the initial form of a typical set of standards data is shown in Table 2-13. The data presented is in table form and represents the standards for a HUFFORD A-12 extrusion stretch forming press, Convair machine code 8030. In this case the total standard hours are made up of two basic items, machine setup time (one increment per job or per die change), and machine run time (one increment per part for preforming and one for finish forming). The run time increments are a function of the overall part length.

The development of the standard hour equations involved acquiring the general standards data and deriving an equation for each manufacturing operation based on the characteristic process and part parameters. For the example standards data (Table 2-13) a general equation would take the form:

$$\text{STDHR} \quad = \quad 0.52 + N * (f_1 (L) + f_2 (L)) \qquad \text{(hours}$$

where

0.52 represents the setup time (constant per job)

N represents the total number of parts to be produced

f_1 (L) represents preform time as a function of part length

f_2 (L) represents finish form time as a function of part length

Table 2-13. Example of Standards Data for Stretch Forming Press as Used by the Industrial Engineering Department

PRESS, EXTRUSION STRETCH FORM

SETUP: 0.52 (ONCE PER DIE CHANGE) MACHINE CODE: 8030

PREFORM

LENGTH cm	0-38	39-76	77-114	115-152	153-191	192-229	230-267
(in.)	(0-15)	(16-30)	(31-45)	(46-60)	(61-75)	(76-90)	(91-105)
STD. HR	0.0255	0.0285	0.0315	0.0345	0.0375	0.0405	0.0435
LENGTH cm	268-305	306-343	344-381	382-419	420-457	458-495	496-533
(in.)	(106-120)	(121-135)	(136-150)	(151-165)	(166-180)	(181-195)	(196-210)
STD. HR	0.0465	0.0495	0.0525	0.0555	0.0585	0.0615	0.0645

FINISH FORM

LENGTH cm	0-38	39-76	77-114	115-152	153-191	192-229	230-267
(in.)	(0-15)	(16-30)	(31-45)	(46-60)	(61-75)	(76-90)	(91-105)
STD. HR	0.0595	0.0625	0.0655	0.0685	0.0715	0.0745	0.0775
LENGTH cm	268-305	306-343	344-381	382-419	420-457	458-495	496-533
(in.)	(106-120)	(121-135)	(136-150)	(151-165)	(166-180)	(181-195)	(196-210)
STD.HR	0.0805	0.0835	0.0865	0.0895	0.0925	0.0955	0.0985

NOTE: LENGTH IN INCHES IS BASED UPON THE BILL OF MATERIAL LENGTH OF PART.
ALL VALUES INCLUDE STOCK ALLOWANCE FOR VISE JAWS

The function of length $f_1(L)$ and $f_2(L)$ are determined by curve fitting the data in the standards table. In this case a linear curve fit is sufficient and the functions resulting are:

$$f_1(L) = 0.0002 * L + 0.058$$

$$f_2(L) = 0.0002 * L + 0.024$$

The resulting standard hour equation for this particular press forming operation then simplifies to:

$$STDHR = 0.52 + N * (0.004 * L + 0.082)$$

A summary of the manufacturing and assembly operations currently represented by equations in the program is presented in Table 2-14.

The standards data are usually derived for aluminum only. To apply the data to additional materials, material complexity factors are utilized. The material complexity factors account for the difference in manufacturing time requirements for performing identical tasks or operations on different materials. These factors are typically required only for those manufacturing operations associated with material removal such as drilling, milling, routing, burring, and cutting.

Table 2-14. Library of Manufacturing and Assembly Operations Currently Available

Boron epoxy layup	Graphite epoxy quality control	Press forming
Boron epoxy quality control	Heat treatment and straightening	Reaming or tapping
Clamping	Identification	Sawing, cutting
Cleaning, degreasing	Inspection (general)	Securing
Cleanup (of holes after drilling)	Inspection for assembly	Setup for assembly
Disassembly (removing clamps for cleaning)	Layout part (sheet)	Shearing
Drilling (general)	Layout holes (sheet)	Spray painting
Drilling for assembly	Layout part (machine shop)	Stretch forming
Edge burring	Layout holes (machine shop)	Surface treatment
Edge routing	Milling (chemical)	Turning (lathe)
Graphite epoxy layup	Milling (general)	Welding, brazing

Operations that usually do not require complexity factors are cleaning, layout, identification, painting, etc. Standard hours for operations performed on steel, titanium, and boron-aluminum are derived in the program using the material complexity factor approach. Table 2-15 summarizes the values for the factors and the processes for which they are applied. Aluminum represents the baseline of 1.0.

Table 2-15. Summary of Material Complexity Factors Applied in the Computation of Standard Hours

	Steel	Titanium	Boron-Aluminum
Burring	3.8	5.0	6.0
Drilling	3.8	5.0	1.2
Forming	3.8	8.0	10.0
Milling	3.8	4.2	1.26
Reaming	3.8	4.0	1.2
Routing	3.8	5.0	6.0
Sawing	3.8	1.1	1.5
Shearing	3.8	1.1	1.5
Turning	3.8	4.2	1.26

Analyses of fabrication processes with advanced composites are handled by assuming three material forms. These are boron-epoxy and graphite-epoxy layups from prepreg tape, and boron-aluminum sheet stock. In general, the advanced composite configurations are assumed to be comprised of the same detail parts, performing the same structural function, as the equivalent metallic configuration.

Boron-epoxy and graphite-epoxy parts are assumed to be layed up to finished form and cured, then bonded into final assembly form. Layup times are computed on the basis of actual hours per unit part weight and per unit part size for boron-epoxy, and on the basis of actual hours per unit part weight for graphite-epoxy. Quality control hours during layup and cure are computed based on hours per unit part size for boron-epoxy, and hours per unit part weight for graphite-epoxy. A realization factor of 1.0 is associated with composite fabrication processes.

A study was made of current data related to advanced composite fabrication operations. As a result of this study it was found that a thorough treatment of each operation in the layup and cure sequence would not be useful. This results from the very limited degree of breakdown of the data that is available, and the fact that much of the data corresponds to the fabrication of only a few actual parts, many of which are relatively simple and physically small. In other words, the data that is presently available is not really representative of a production situation involving actual aircraft components, and is, for the time being, treated in a more simplified manner.

The actual hour computation procedure that is in use for boron-epoxy and graphite-epoxy assumes that the sequence of processes can be combined into two, layup and quality control. These are treated on the basis of hours per unit size and weight. Expressions for actual hours are:

(4.18 * FFF * ACWT * CAREA) ** .5	boron-epoxy layup
.220 * CAREA	boron-epoxy quality control
9.6 * ACWT	graphite-epoxy layup
1.2 * ACWT	graphite-epoxy quality control

where

FFF is a factor corresponding to part configuration

ACWT is the part actual weight

CAREA is the characteristics part area

Boron-aluminum is assumed to be in the form of sheet stock. Standard hours are computed using the ordinary equations times a material complexity factor. These factors were summarized in Table 2-15.

2.3.1.4 Rate Data: Labor, Overhead, and Realization. In the program standard hours are computed as an intermediate step in the process of deriving actual labor hours. The conversion is accomplished by making use of the realization factor, a measured value representing shop efficiency as discussed below. The equations for actual labor hours take the following form:

Actual Labor Hours = Standard Hours/Realization

Labor and overhead rates are used in the program to calculate labor and overhead costs, based on the number of actual labor hours required for each manufacturing and assembly process. Labor rates reflect the wages paid directly to the individual employees for each hour of clock time. The rates do not include fringe benefits or company contributions to retirement, social security, state unemployment, etc., which are considered part of the overhead cost. Also included as part of overhead are indirect labor costs, maintenance, supplies, taxes, insurance, depreciation, etc.

Labor rates are largely uncontrollable by management, being a function instead of union/management agreements and reflecting current labor supply and demand, general economic conditions, and inflation. Labor rates are a function of time and are readily predictable for the near future.

The overhead ratio is the ratio of overhead costs to direct labor costs. It is established based on historical accounting records, and is, in turn, often used by estimating departments. In the program, the overhead ratio is used to determine the overhead costs corresponding to the calculated labor costs where:

Overhead Cost = Labor Cost * Overhead Ratio

Realization is a measure of shop efficiency, and as such, it varies from department to department and from day to day within a department. Realization data for the various departments involved in production tasks at the San Diego operation has been collected, studied, and adapted for use with the program. Realizations can be specified either as a constant average value or as a time dependent equation. Some of the factors affecting realization are:

a. Worker personal needs.

b. Rest periods.

c. Inaccurate planning of the task.

d. Change in procedure, machines, or tools without corresponding change in manhour estimates.

e. Machine breakdown.

f. Tool breakage and part spoilage.

g. Availability of previous setups.

h. Use of special supervision.

i. Ability and effort level of individuals assigned the task.

Labor and overhead costs are computed directly for the first unit. A learning curve approach is applied to first unit costs to derive the cost of any subsequent unit or production lot. Labor (and overhead) costs are generated at the detail part level. For each manufacturing or assembly process specified for a given part, a value for standard hours, actual labor hours, labor cost, and overhead cost is computed. These are summed to obtain total costs for a given part, subassembly, assembly, etc.

Manufacturing and assembly processes have been divided into three groups: basic factory, quality control, and assembly. For each there is available in the program a corresponding labor rate that is computed from a base value as a function of time. Also available in the same manner are values for overhead ratio and realization factor.

For each, average industry values for 1970 are used as the base. Rate data for any year is computed by assuming a constant fractional annual rate of change. Any of the internal values for rate or annual rate of change may be overridden by direct input. In the absence of an input, values are computed. A summary of the rate data internal to the program is presented in Table 2-16.

Table 2-16. Summary of Internal Program Rate Data

Description 1970 Base Year	Rate Data Fortran Name	Rate Data Value	Annual Rate of Change Fortran Name	Annual Rate of Change Value
Factory Labor Rate ($/hr)	FRATE	3.64	CLAR	0.055
Quality Control Labor Rate ($/hr)	QRATE	4.06	CLAR	0.055
Assembly Labor Rate ($/hr)	ARATE	3.48	CLAR	0.055
Overhead Ratio	VRATE	1.80	COVR	0.02
Realization Factor	REFCT	0.15	CRE	0.01

2.3.2 MATERIAL COST. Material costs are computed based on the material type (aluminum, steel, etc.), material form (sheet, plate, bar, etc.), and the raw material purchase weight. The actual calculation of material cost takes the form:

MATCOS = AMUV * COSWT * MAWT

where

MATCOS is the material cost

AMUV is the manufacturing usage variance factor explained below

COSWT is the material unit cost

MAWT is the raw material purchase weight

The computation of material costs requires the derivation of a material unit cost (COSWT) and the definition of a manufacturing usage variance factor (AMUV). The computation of the material purchase weight (MAWT) is done during the weight analysis portion of the program.

The material unit cost is, in general, a function of the material type, form, quantity of material bought, and special feature requirements such as special lengths, widths, thicknesses, alloys, tempers, tolerances, and marking. Computation of the material unit cost can be summarized as follows: a base price is computed as a function of material type and form; the base price is adjusted to account for the quantity buy price differential; the prices of appropriate special feature extra cost items are computed and summed to derive a total special feature penalty cost; a total material unit cost

is determined by summing the adjusted base price and the special feature penalty cost; and finally, the resultant value for material unit cost is adjusted, if necessary, to correspond to dollars for the specified reference year.

Material type is specified by input of a value for MATLID, which represents the component structural material. The materials currently available in the program are:

MATLID = 1 Aluminum	MATLID = 4 Boron-Epoxy
MATLID = 2 Steel	MATLID = 5 Boron-Aluminum
MATLID = 3 Titanium	MATLID = 6 Graphite-Epoxy

Material form is specified by defining a value for KEY in the parts definition subroutines. Each detail part is assumed manufactured from and is thus associated with one of the following material forms:

KEY = 1 FASTENER

2 HONEYCOMB

3 FOIL, SHEET, PLATE

4 WIRE, ROD, BAR

5 EXTRUSION

For a given material type and form, a value for the total quantity of material purchased is computed by summing the values for raw material purchase weight for each detail part. A system of arrays is defined to categorize materials by type, form, and stock dimensions. The value derived for material weight for each detail part is added to one of the array elements as it is computed. After material purchase weights have been computed for all detail parts, the system of arrays is multiplied by the number of shipsets to be produced. By this means a total quantity of required material is available for computing material costs as a function of quantity bought.

By specifying the material type (MATLID) the program is directed to one of two fundamental areas of material cost computation. The first encompasses the metallic materials aluminum, steel, and titanium, and the second encompasses the advanced composites boron-epoxy, boron-aluminum, and graphite-epoxy. The primary difference in methodologies reflected by the two areas is due to the assumptions made with respect to material form. For the metallics a material form (KEY) is specified by the parts prediction routines, and the unit material costs are computed as a function of the form. For the advanced composites a material form is assumed, and all parts are considered to be fabricated of the assumed material form. The assumed material forms are 7.62-cm (3-in.) wide prepreg tape for boron-epoxy and graphite-epoxy, and cured sheet for boron-aluminum.

Unit costs for metallic materials are computed as a function of both material type and form. Price data for various materials and material forms were collected and curve fit as a function of nominal material stock sizes. Table 2-17 illustrates a typical base price schedule for alloy steel plate between 0.635 cm (0.25 in.) and 15.24 cm (6.0 in.) thick. The resulting equation for this particular material is:

Table 2-17. Part of Typical Material Price Schedule for Alloy Steel Plate (1970 Data)

Thickness (cm)	Thickness (in.)	E4340, AMS-6359 Hot Rolled Annealed ($/100 kg)	($/100 lb)
0.635	(0.250)	99.00	(45.00)
0.953	(0.375)	97.68	(44.40)
1.270	(0.500)	97.13	(44.15)
1.588	(0.625)	97.35	(44.25)
1.905	(0.750)	97.02	(44.10)
2.540	(1.000)	96.69	(43.95)
3.175	(1.250)	96.80	(44.00)
3.810	(1.500)	96.80	(44.00)
4.445	(1.750)	98.01	(44.55)
5.080	(2.000)	98.01	(44.55)
5.715	(2.250)	104.28	(47.40)
6.350	(2.500)	104.28	(47.40)
6.985	(2.750)	104.28	(47.40)
7.620	(3.000)	104.28	(47.40)
8.890	(3.500)	104.28	(47.40)
10.160	(4.000)	104.28	(47.40)
11.430	(4.500)	104.28	(47.40)
15.240	(6.000)	104.28	(47.40)

$$PBASE = 0.006 * THK + 0.439$$

where

PBASE is the unit base price

THK is the material thickness

Thus, by specifying MATLID = 2, KEY = 3, and THK equal to some characteristic or computed thickness, the program calculates a unit base price for the required size of alloy steel plate. In a similar manner, equations were derived to provide a means of computing base price data for the various forms of aluminum, steel, and titanium.

For some combinations of material type and form, such as titanium extrusions, specific price data was not available. For these cases a characteristic material base price was established, as MBASE = 8.50 for titanium. The specified material was then analyzed in terms of the equivalent aluminum material form (aluminum extrusions), and the resulting value of PBASE derived for the equivalent aluminum form was ratioed using an aluminum base price (ALBASE = 0.80) and the specified material base price (MBASE). Table 2-18 is a summary of the values of MBASE currently being used in the program. Table 2-19 is a summary of the material type and forms currently available in the program.

Table 2-18. Summary of Values for the Characteristic Material Base Price Currently in Use in the Program

Aluminum	ALBASE = 0.80
Titanium	MBASE = 8.50
Alloy steel	MBASE = 0.40

A price differential based on the quantity of material purchased is computed and

Table 2-19. Summary of Material Type and Forms Currently Available in the COSTMA Subroutine

				Al	Steel	Ti
KEY	=	1	fastener	●	●	●
		2	honeycomb	●	o	o
		3	foil, sheet, plate	●	●	●
		4	wire, rod, bar	●	●	●
		5	extrusion	●	o	o
		6	tubing	To be added at a future date		
		7	forging			
		8	casting			

● direct material price data available

o ratioed material price data available

used to adjust the unit base price. Equations defining the price differential were derived by curve fitting quantity purchased versus unit cost data. An example of data typical of the type utilized is presented in Table 2-20.

A cost penalty is determined for required extra cost special features. Equations for each of typical extra cost items have been generated from material vendor pricing handbooks. These equations include costs for protective coatings, identification, mechanical testing, packing, shipping, etc. (Table 2-21). The cost penalty is added to the adjusted material unit cost to provide a total material unit cost.

Table 2-20. Example of the Quantity Buy Price Differential for Aluminum Plate

Quantity per Item		Extra	
kg	(lb)	\$/kg	(\$/lb)
13,636 and over	(30,000 and over)	Base	(Base)
13,635 - 9,091	(29,999 - 20,000)	0.022	(0.010)
9,090 - 4,545	(19,999 - 10,000)	0.044	(0.020)
4,544 - 3,636	(9,999 - 8,000)	0.110	(0.050)
3,635 - 1,818	(7,999 - 4,000)	0.154	(0.070)
1,817 - 1,364	(3,999 - 3,000)	0.275	(0.125)
1,363 - 909	(2,999 - 2,000)	0.627	(0.285)
908 - 682	(1,999 - 1,500)	0.990	(0.450)
681 - 455	(1,499 - 1,000)	1.705	(0.775)

Table 2-21. Summary of Extra Cost Items Available for Aluminum Plate

PRICING CHECK LIST

The following General Extras apply to sheet and plate products.

PLATE	
ACTUAL PIECE COUNT	PACKING
ALLOYS AND SPECIAL EXTRAS	PACKING PER MIL-STD 649-SHEET AND PLATE
CIRCLES	PROTECTIVE TAPE
CONVERSION COATINGS	QUANTITY
EXACT QUANTITY	TEST MATERIAL SAMPLES
IDENTIFICATION MARKING-STANDARD	TOLERANCES
IDENTIFICATION MARKING-SPECIAL	DIAMETER
INTERLEAVING AND OILING	FLATNESS
LENGTHS, LONG	LENGTH
LENGTHS, SHORT	THICKNESS
MACHINED SURFACE (TWO SIDES)	WIDTH
MECHANICAL TESTING	ULTRASONIC INSPECTION

Material unit costs for advanced composites are computed directly as a function of the reference year for boron-epoxy and boron-aluminum and as a combined function of the average single-ply thickness and reference year for graphite-epoxy. The equations were derived by curve-fitting actual and projected cost data acquired from the Convair Aerospace Advanced Composites Laboratory and from typical material vendors. The equations are:

$$P = 225 - 16.5 * (YR - 70) \qquad \text{Boron-Epoxy}$$

$$P = 425 - 22.5 * (YR - 70) \qquad \text{Boron-Aluminum}$$

$$P = 115/(PLYT - 1.111) - 9.3 * (YR - 70) + 89 \qquad \text{Graphite-Epoxy}$$

where

P is the material unit cost

YR is the dollar reference year

PLYT is the average single-ply thickness

A plot of material cost versus year for boron-epoxy and boron-aluminum is presented in Figure 2-37. Graphite-epoxy costs versus year for various single-ply thicknesses are presented in Figure 2-38.

The value for average single-ply thickness (PLYT) is computed from material thickness. A range of from 0.013 to 0.051 cm (5 to 20 mils) is allowed. If the computed

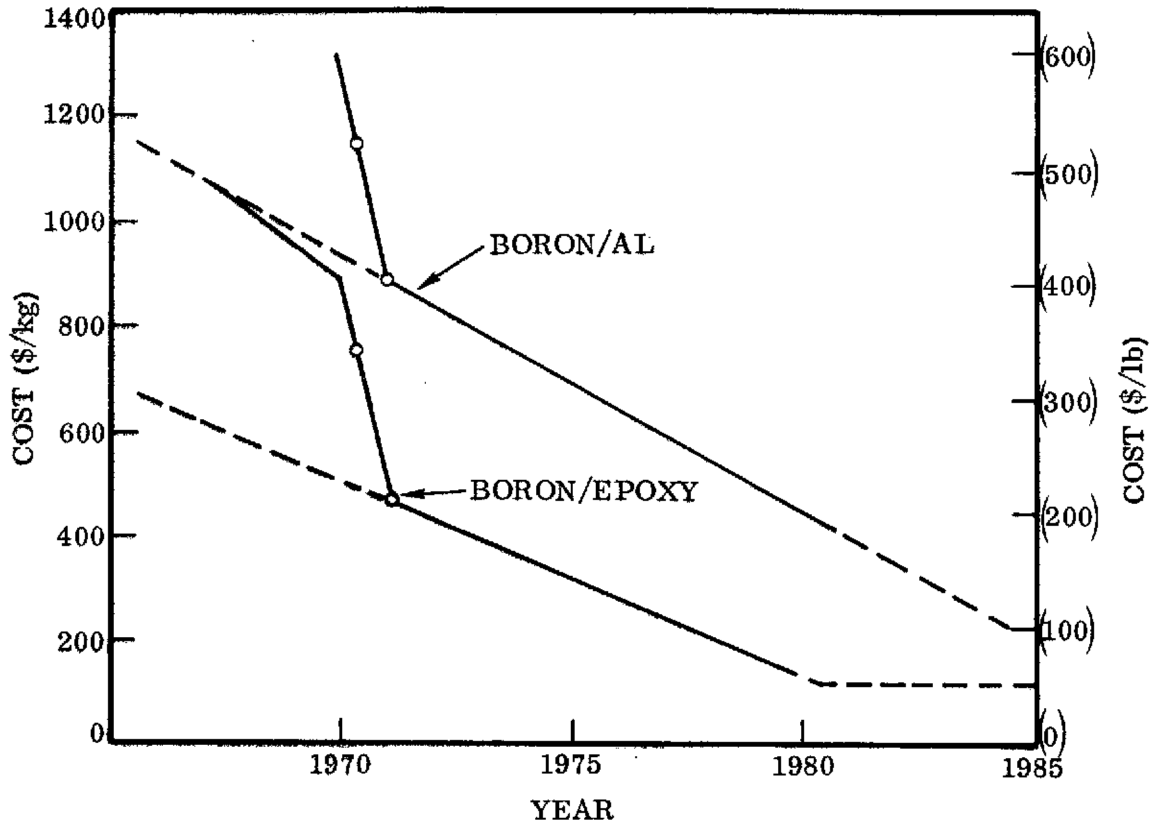


Figure 2-37. Projected Raw Material Costs for Boron-Aluminum and Boron-Epoxy

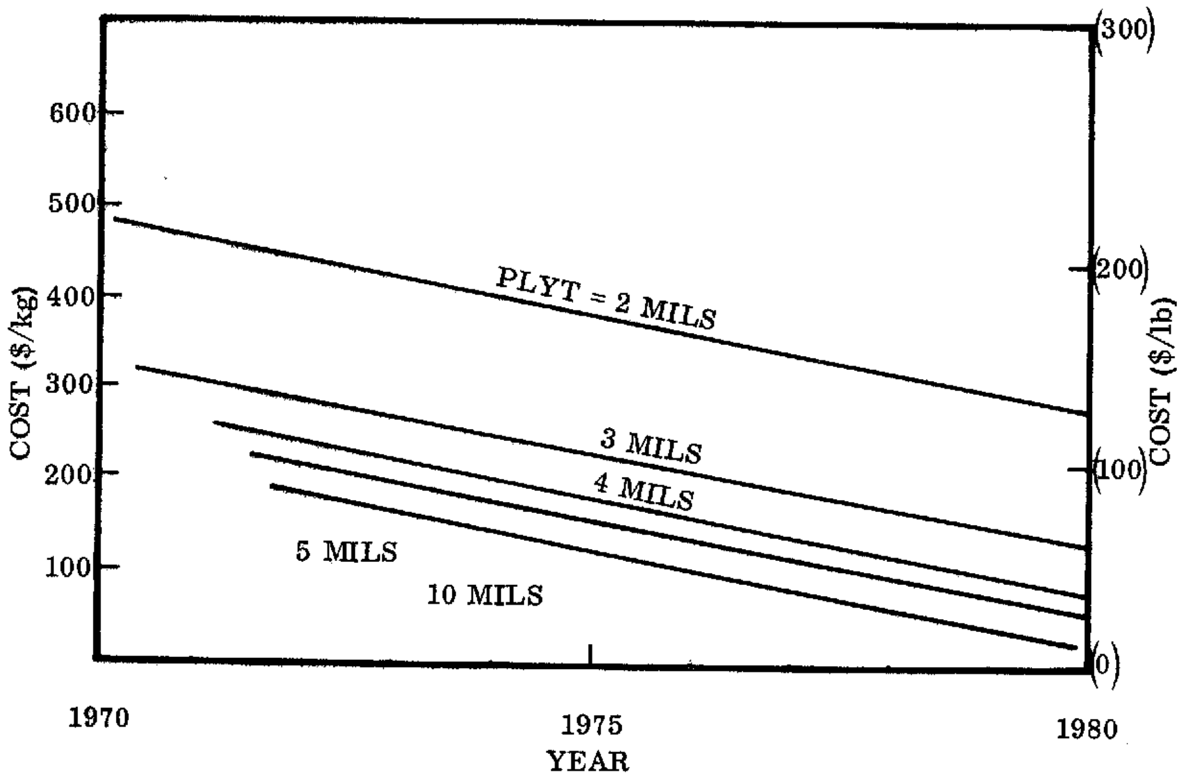


Figure 2-38. Projected Raw Material Costs for Graphite-Epoxy

value is outside this range, a value of 0.013 cm (5 mils) is set. A minimum value for each of the material unit costs has also been fixed. The unit costs, which override smaller computed values, are: $110/kb ($50/lb), boron-epoxy; $220/kg ($100/lb), boron-aluminum; and $22/kg ($10/lb), graphite-epoxy. It was determined that these were the minimum material prices that would be achieved in the foreseeable future. A price differential based on the total material quantity purchased was established as follows: a 10% penalty was added to the unit cost for purchases of 4545 to 45,455 kg (10,000 to 100,000 lb), 20% for purchases of 455 to 4545 kg (1,000 to 10,000 lb), 40% for purchases of 45 to 455 kg (100 to 1,000 lb), and 60% for purchases of less than 45 kg (100 lb).

While the unit material cost for advanced composites is computed as a function of the dollar reference year directly, the unit cost for metallics must be adjusted to the reference year. An adjustment is made assuming a constant annual rate of inflation. A value for the annual rate of inflation of material costs (AINFL) may be input directly by the user; in the absence of an input a nominal value of 0.03 is assumed.

The manufacturing usage variance factor AMUV is the ratio of the actual amount of material purchased to the sum of the engineering net bill of materials plus the planning allowances for manufacturing. The factor is, in general, a function of material type (particularly in the case of advanced composites) and past material handling experience. The factor results from material and part overbuying to account for normal indirect material losses during the manufacturing phase of production. These losses include material and part spoilage, duplication, substitution, changes, waste, etc. These losses are to be differentiated from those resulting directly from manufacturing, such as trimming, routing, and milling, which are accounted for in the derivation of the material purchase weight.

The actual value for the manufacturing usage variance factor is determined by accounting procedures. Data from several past programs are presented in Table 2-22. A nominal value of 1.10 is currently in use by the program for all material forms. This represents a 10% overbuy and is a fairly good average value for typical metallic aircraft construction. However, it is somewhat high for production involving the use of advanced composite materials.

2.3.3 ENGINEERING COSTS

2.3.3.1 Engineering Cost Derivation. Engineering costs are computed by deriving the number of engineering manhours required and multiplying this by a composite engineering labor rate. Engineering hours are computed as initial engineering hours – those hours utilized by the time the first airframe has been completed, and sustaining engineering hours – those hours occurring after the first airframe has been completed.

The actual computation of initial and sustaining engineering hours has as its basis equations developed by Levinson and Barro (Reference 13). For this reason their

Table 2-22. Typical Manufacturing Usage Variance Factors for a Past Commercial Transport Program

CONTRACT LOT NO.	ACTUAL MATERIAL COSTS A (millions of $)	ORIGINAL ESTIMATED MATERIAL COSTS E (millions of $)	PERCENT VARIANCE $\left(\frac{A - E}{A}\right) * 100$ (percent)	MANUFACTURING USAGE VARIANCE FACTOR
1	40.65	34.74	17.0 %	1.170
2	4.61	4.25	8.5	1.085
3	16.67	14.39	13.8	1.138
4	22.69	21.40	6.1	1.061
5	16.28	15.84	2.8	1.028
6	66.50	62.15	7.0	1.070
7	10.22	9.84	3.9	1.039
8	68.71	61.94	10.9	1.109

definition of the engineering task was used. Engineering, then, was defined as including the following: design and integration studies, engineering for wind tunnel models, mockup and engine testing, test engineering, laboratory work on subsystems and static test items, development testing, board hours, release and maintenance of drawings, specifications, shop and vendor liaison, analysis and incorporation of changes, material and process specifications, and reliability. Hours not considered as engineering include those associated with flight test, planning, ground handling equipment, spares, mobile training units, and publications. Also not included as part of engineering cost are travel and computer time.

The basic equation used for initial engineering hours is:

ENGRHR = 8.0 * (VELALT ** 0.55) * (FP ** 0.88) * CONFAK

where

VELALT is the maximum aircraft speed at cruise altitude

FP is the total aircraft sea level thrust

CONFAK is an engineering configuration complexity factor with a nominal value of 1.0

The total number of initial engineering hours computed is broken down and distributed among the various engineering disciplines based on percentages derived from studies of historical data. There are two basic breakdowns, one corresponding to a typical subsonic or transonic transport type aircraft and one corresponding to a typical high performance military type aircraft. Maximum Mach number is used to differentiate between the two breakdowns with a Mach number of 1.1 or greater corresponding to the military type aircraft.

The value for total initial engineering hours that is output is not the value that is computed directly, but is instead a value found by summing the hours for all of the various engineering disciplines. It is hoped that at a future date each area of engineering can be looked at individually, and that for each, methods can be developed to derive hours directly by means of empirical relationships. As these methods are developed, the percentage based computation for a given discipline can be easily replaced by a direct computation, and the new equations will thus be represented in the final output value for initial engineering hours. In this way, an empirically based routine can be built up bit by bit while retaining the capability of generating a complete output during the development.

The initial engineering hour breakdowns into the various design and support disciplines are shown in Table 2-23. The fractions shown for each discipline were derived by studying actual hour data for the F-102, F-106, F-111, and 880/990 aircraft, and estimated data for the VS(X). Other detailed data for current aircraft were not available. It should be noted that the fractional data shown are averaged values for several

Table 2-23. Initial Engineering Hour Breakdown as a Fraction of Total Design Hours (DESHR) where DESHR = 0.54 * ENGRHR for Mach < 1.1 and DESHR = 0.48 * ENGRHR for Mach ≥ 1.1

	Mach < 1.1		Mach ≥ 1.1	
Structural/Mechanical Design		0.86		0.82
Wing	0.14		0.12	
Tail	0.07		0.04	
Body	0.15		0.20	
Furnishings	0.14		0.08	
Gear	0.04		0.04	
Propulsions	0.12		0.10	
Controls	0.09		0.09	
Environmental Control	0.05		0.05	
Hydraulic/Pneumatic	0.05		0.06	
Reliability	0.01		0.01	
Armament	0		0.03	
Electrical/Electronic Design		0.14		0.18
Design Support		0.25		0.25
Lines/Loft	0.15		0.15	
Drafting/Isometrics	0.02		0.02	
Checking/Release	0.06		0.06	
Liaison/Support Design	0.02		0.02	
Technical Support		0.60		0.60
Stress	0.16		0.16	
Weights	0.06		0.06	
Aero	0.05		0.05	
Dynamics	0.08		0.08	
Thermo	0.08		0.08	
Test Lab	0.10		0.10	
Electrical	0.06		0.06	
Staff	0.01		0.01	
Predesign		0.02		0.02
Standards/Specifications/Publications		0.06		0.06

aircraft; consequently, the sum of the fractions do not necessarily yield exactly 100% of the computed value for total initial engineering hours. The value for total hours that is utilized is the sum of the hours for the various disciplines, not the directly computed value.

Sustaining engineering hours are computed and output based on the total number of shipsets. Levenson and Barro found the sustaining hours were not systematically a function of aircraft physical or performance characteristics, and hence could be represented by the equation (Reference 13):

SUSEHR = ENGRHR * (SHPSET ** 0.20 - 1.0)

where

SHPSET is the total number of aircraft shipsets produced.

Sustaining engineering for a given production lot may be computed from:

SUSEH(N) = ENGRHR * (SHP(N) ** 0.20 - SHP(M) ** 0.20)

where

M+1 is the ship number of the first ship in the lot

and

N is the ship number of the last ship in the lot.

Engineering hours are assumed to be a function of aircraft performance and not directly a function of material or type of construction. Engineering hours for advanced composite structures, in particular, are assumed to be initially the same as for aluminum structures. However, the number of hours is expected to decrease later with learning (Reference 14). This assumption is based on the fact that composite structures are charcterized by fewer parts but by a higher degree of learning.

In general, adjustments to engineering hours to reflect unusual material or structural arrangements can be handled through the use of the engineering configuration complexity factor CONFAK. This factor has a nominal value of 1.0, which can be changed at the users option by direct input.

2.3.3.2 Engineering Labor Rate. Engineering labor rate may be input directly as a user option. If a value is not input a rate is computed based on the reference year. A single rate is applied to all engineering tasks.

To derive the equations for engineering labor rate, the rate data from several literature sources were plotted versus time (Figure 2-39). The data utilized were a composite rate composed of direct, indirect, general and administrative, and allocations charges. An average rate was derived for each of the years plotted and a smooth curve was faired through the average values in three segments. Equations were derived to fit each segment as a function of year, resulting in the following:

$YR < 68$	ERATE = .5129 * YR - 22.308
$68 \leq YR < 70$	ERATE = 2 * YR - 123
$YR \geq 70$	ERATE = 17 * (1 + EIFAC) ** (YR - 70)

where

EIFAC is an annual rate of inflation of the engineering labor rate, which has a nominal value of 0.06 but may be input as an option.

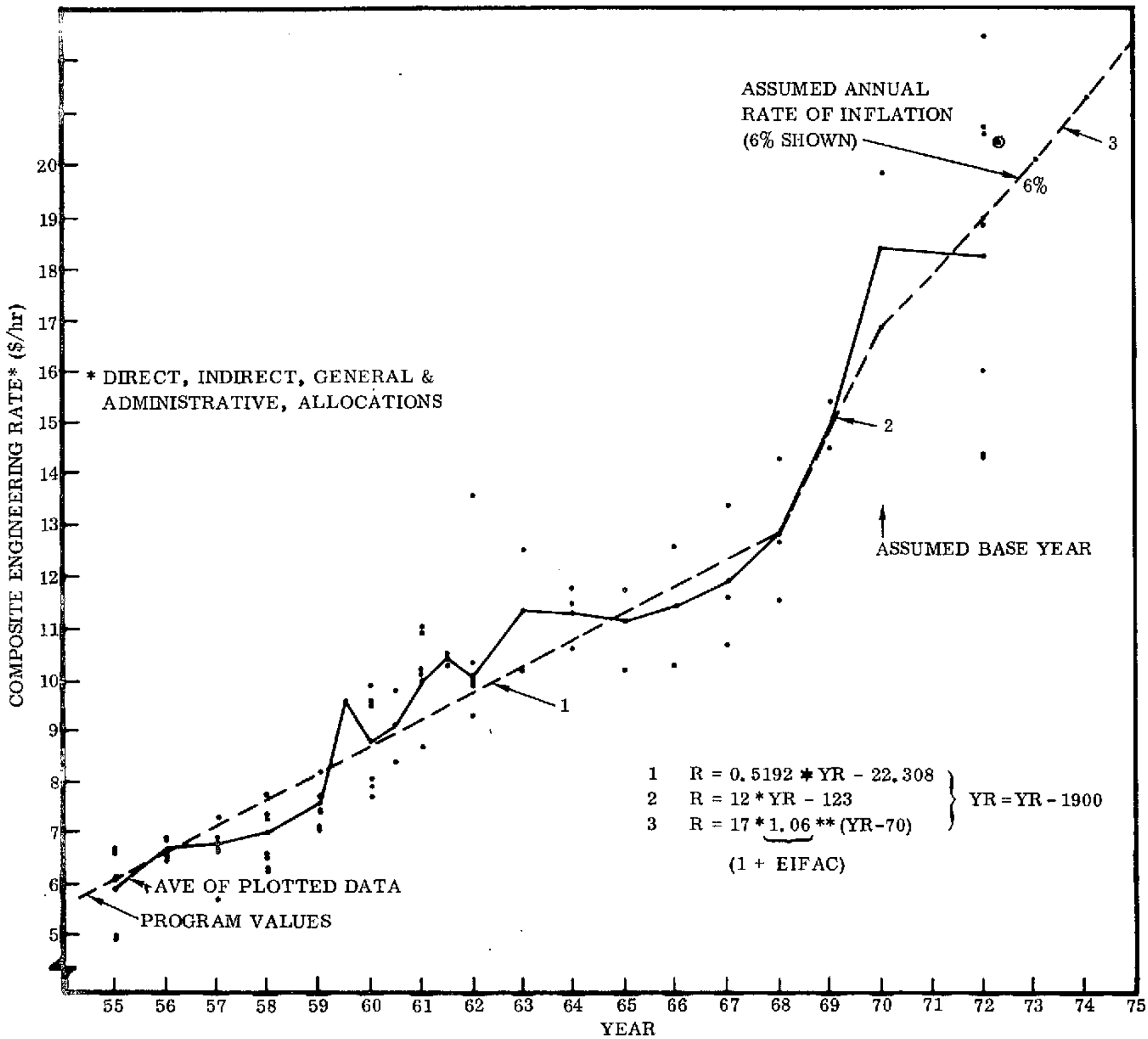


Figure 2-39. Engineering Labor Rate Versus Year

2.3.4 TOOLING COSTS

2.3.4.1 Tooling Cost Derivation. Tooling costs are comprised of three primary elements. They are: basic tooling, which is the first level of tooling designed to support the initial production lot at the initial production rate; rate tooling, which is the second level of tooling established to support the remainder of the production schedule at the maximum production rate; and sustaining tooling, which is the tooling effort required to support the entire production schedule by providing for tool maintenance and producibility charges.

Each of the three tooling elements can, in turn, be broken down into manufacturing, engineering, and materials. Tool manufacturing includes the following: tooling

Table 2-24. Summary of the Tooling Cost Breakdown

Non Recurring Tooling
Basic Tooling
Manufacturing
Engineering
Material
Rate Tooling
Manufacturing
Engineering
Material
Recurring Tooling
Sustaining Tooling
Manufacturing
Engineering
Material

machine shop, template shop, plastic pattern shop, foundry, jigs and fixtures, tool and die, form blocks, and plastics. Tool engineering includes tool design, tool and operations planning, tool project engineering, numerical control programming, tool liaison, production control, and proofing. Tooling materials include materials and graphic reproduction support. A summary of the tooling cost breakdown is listed in Table 2-24.

Tooling costs are computed as a function of the number of basic tooling manufacturing hours (BTMH), initial and sustaining production rates (RI and RS), and tool manufacturing and engineering labor rates (TRATEM and TRATEE). Following are the equations used (References 13 and 15):

Basic Tooling Costs

BMFGS = 1.00 * BTMH * TRATEM * RI ** .4

BENGRS = .40 * BTMH * TRATEE * RI ** .4

BMATLS = 1.20 * BTMH * RI ** .4

Rate Tooling Costs

RMFGS = .10 * BTMH * TRATEM * (RS ** .4 - RI ** .4)

RENGRS = .015 * BTMH * TRATEE * (RS ** .4 - RI ** .4)

RMATLS = .120 * BTMH * (RS ** .4 - RI ** .4)

Sustaining Tooling Costs

SMFGS = 1.00 * BTMH * SUMFAC * TRATEM * RS ** .4

SENGRS = .50 * BTMH * SUMFAC * TRATEE * RS ** .4

SMATLS = .90 * BTMH * SUMFAC * RS ** .4

where

SUMFAC is a production rate factor discussed below.

Basic tooling manufacturing hours are computed based on the number of dissimilar parts to be produced (DISPRT), the average number of tools required per dissimilar part (TOOLPP), and the average number of hours required to produce each tool (HRPTOO).

$$BTMH = CONFAC * DISPRT * TOOLPP * HRPTOO$$

where

CONFAC is a tooling configuration complexity factor discussed below.

A value for total number of dissimilar parts (DISPRT) can be input directly or in the absence of an input is calculated from the following (Reference 16):

$$DISPRT = 16.53 * AMPRWT ** 0.728$$

where

AMPRWT is the AMPR weight of the aircraft.

The equation is illustrated in Figure 2-40. It is hoped that eventually the number of dissimilar parts can be derived directly from a parts count made in the parts definition portions of the program, rather than using a statistical derivation driven by weight.

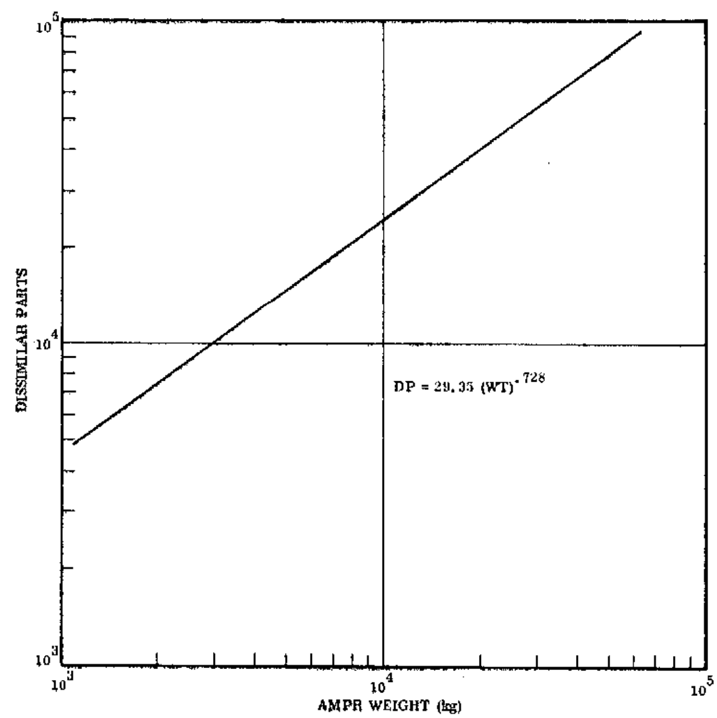


Figure 2-40. Number of Dissimilar Parts Versus AMPR Weight for the Complete Airframe

A plot of total tools versus the number of dissimilar parts is shown in Figure 2-41. A nominal value of 1.8 is used for the average number of tools required per dissimilar part (TOOLPP) in the absence of a direct input by the user. Figure 2-42 shows typical values of the average number of hours required to produce a tool (HRPTOO) plotted against number of dissimilar parts. A nominal value of 49.0 is used by the program in the absence of a direct input. A summary of the data that was available for the analysis of tooling cost is presented in Table 2-25.

The production rate factor (SUMFAC) represents a tool maintenance fraction, which is a function of the aircraft production rate and the number of shipsets produced. It is computed from the following:

$$SUMFAC = \sum_{i=LOTO}^{LOTS} (NOMO_i * Factor_i)$$

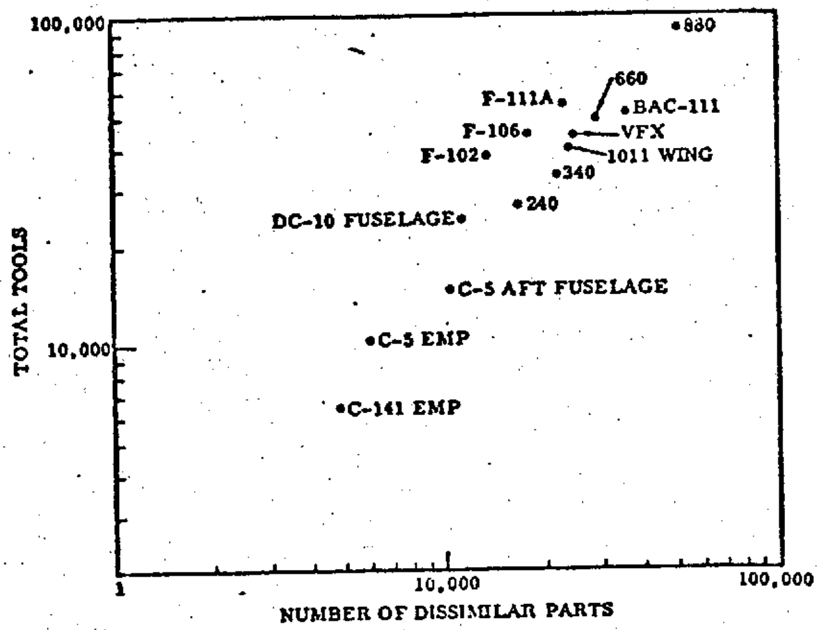


Figure 2-41. Number of Tools Required as a Function of Total Dissimilar Parts

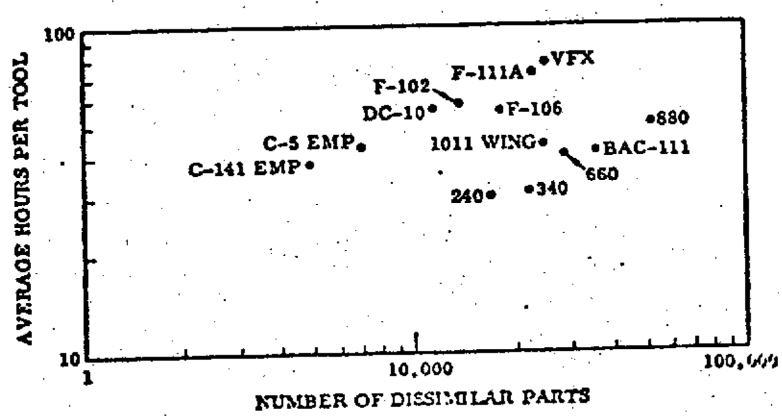


Figure 2-42. Average Number of Tooling Manufacturing Hours Required per Tool

where

$NOMO_i$ is the number of months required to produce the shipsets of LOT_i

$FACTOR_i$ is a factor computed from the curve of Figure 2-43.

The number of months each lot is under production is computed by dividing the shipsets in each lot by the production rate corresponding to that lot. A value for FACTOR is taken from the curve of Figure 2-43 as follows: a value of 0.015 is used for the first lot, or the first 10 ships of the first lot if the total is greater than 10; for each successive lot up to ship number 150, and for the remaining ships of the first lot if the total is greater than 10, a value is computed using:

Table 2-25. Summary of Tooling Cost Data Used in the Analysis

Program	AMPR Weight kg	(lb)	Diss. Parts	Tools/ Part	Total Tools	Average Hr/Tool	Tool Mfg. Hr
A	9,017	(19,838)	16,785	1.51	25,400	29.6	751,734
B	9,851	(21,673)	22,000	1.51	33,200	31.0	1,029,820
C	29,864	(65,700)	51,000	1.77	90,181	50.2	4,526,110
D	39,614	(87,150)			66,154	45.0	2,986,930
E	5,488	(12,074)	13,815	2.62	36,191	58.0	2,099,772
F	6,835	(15,037)	18,166	2.31	42,060	55.7	2,341,320
G	14,923	(32,830)	35,866	1.44	51,751	40.6	2,100,000
H	2,767	(6,087)	4,871	1.30	6,315	38.4	242,363
I	5,381	(11,839)	6,077	1.72	10,439	41.4	432,059
J	19,268	(42,390)	24,020	1.69	40,506	43.8	1,772,730
K	13,000	(28,600)	28,800	1.70	48,960	40.0	1,958,400
L	8,301	(18,263)	10,709	1.36	14,569	31.8	559,440
M	14,795	(32,548)	22,741	2.34	53,000	71.0	3,775,000
N	11,530	(25,365)	24,300	1.7	42,200	77.0	3,250,000
O	15,075	(33,166)	11,367	2.13	24,174	55.0	1,314,467
P	7,045	(15,500)					2,165,600

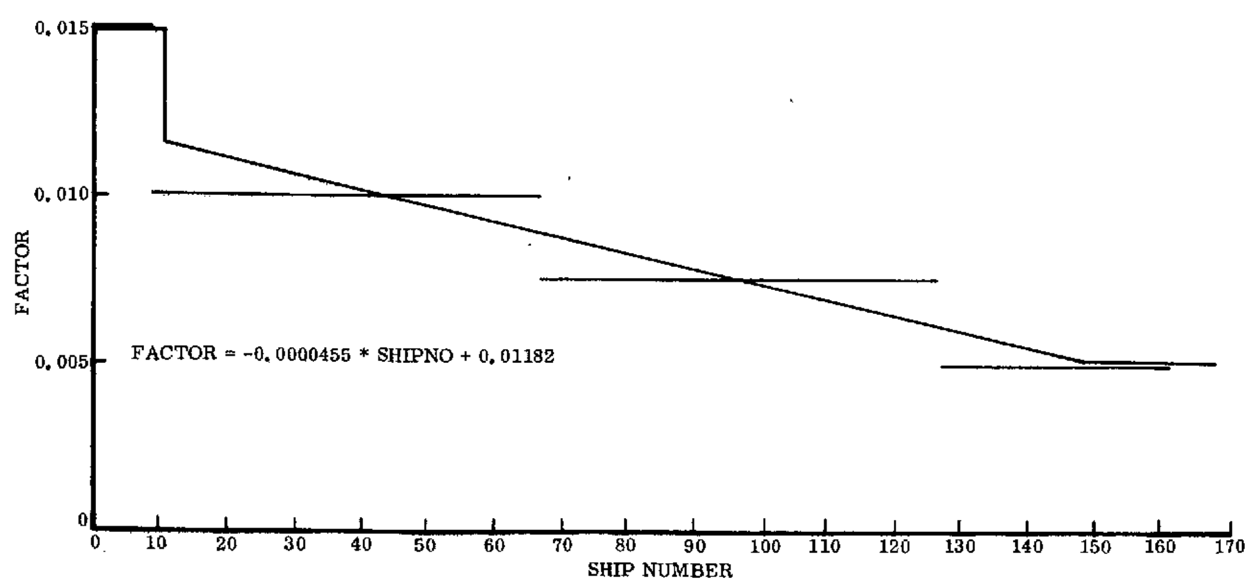


Figure 2-43. Plot of the Tooling Maintenance Factor per Month Versus Ship Number

$$FACTOR_i = -0.0000455 * SHPNO_i + 0.01182$$

where

$SHPNO_i$ is the middle ship number of lot_i

For the remaining lots (above ship number 150) a value of 0.005 is used.

The tooling configuration complexity factor (CONFAC) was designed to account for different materials and structural arrangements. It has a nominal value of 1.0, which is used in the absence of a direct input. Table 2-26 lists some suggested values for the factor. It should be noted that an aircraft constructed of advanced composite materials was assumed to require 70% of the tooling necessary for a comparable metallic version (Reference 14).

Table 2-26. Suggested Input Values for Tooling Configuration Complexity Factor CONFAC

	Metallic	Combination Metallic/Composite	Composite
Simplified Design, Follow-on Subsonic	0.8	0.7	0.5
Regular Subsonic	1.0	0.9	0.7
Complex Subsonic; Simplified Design, Follow-on Supersonic	1.8	1.6	1.2
Regular Supersonic	2.2	1.9	1.5
Complex Supersonic	2.5	2.2	1.8

The initial and sustaining production rates (RI and RS) are given nominal values of 1.0 in the absence of a direct input. The initial production rate (RI) is assumed to encompass the production of the RDT&E (preproduction) and LOT1 ships, and the sustaining rate (RS) is assumed to encompass the remainder, LOT2 through LOT5. A summary of the tooling cost elements as related to the assumed production schedule is illustrated in Figure 2-44.

2.3.4.2 Tooling Labor Rates. Tool engineering and manufacturing labor rates may be input as a user option. If a value for either is not input, a rate is calculated based on the reference year.

To derive equations for tool engineering and manufacturing labor rates, rate data (combined average of engineering and manufacturing) from several literature sources

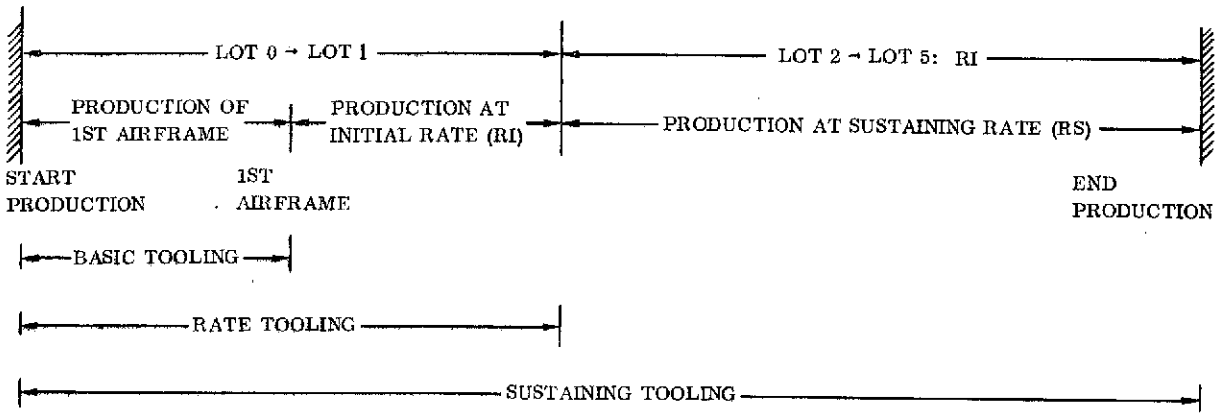


Figure 2-44. Summary of Tooling Cost Elements as Related to the Production Schedule

were plotted versus time (Figure 2-45). The data utilized were a composite rate composed of direct, indirect, general and administrative, and allocations charges. An average rate was derived for each of the years plotted and a smooth curve was faired through the average values in three segments. Equations were derived to fit each segment as a function of year, resulting in the following:

YR < 68	RATE = 0.3846 * YR - 15.1538
68 ≤ YR ≤ 70	RATE = 1.5 * YR - 91
YR > 70	RATE = 14 * (1 + TIFAC) ** (YR - 70)

where

TIFAC is an annual rate of inflation of the tooling labor rate, which has a nominal value of 0.06 but may be input as an option.

The resultant value for labor rate is then adjusted to correspond to either the engineering or manufacturing areas of tooling cost. It was found that tool engineering and tool manufacturing labor rates are usually separated by about 7%. For this reason the average calculated labor rate is increased by 3.5% to derive a tool engineering rate, and decreased by 3.5% to derive a tool manufacturing rate.

TRATEE = 1.035 * RATE

TRATEM = 0.965 * RATE

2.3.5 TOTAL VEHICLE PROGRAM COSTS. Total vehicle program costs are computed based on a cost model that was assembled primarily utilizing the work of Kenyon (Reference 15). The model incorporates a general format similar to that used by Kenyon although equations taken from the referenced literature have been substituted in several places. Where possible values for various cost elements that have been computed elsewhere in the program are brought across. These include first unit

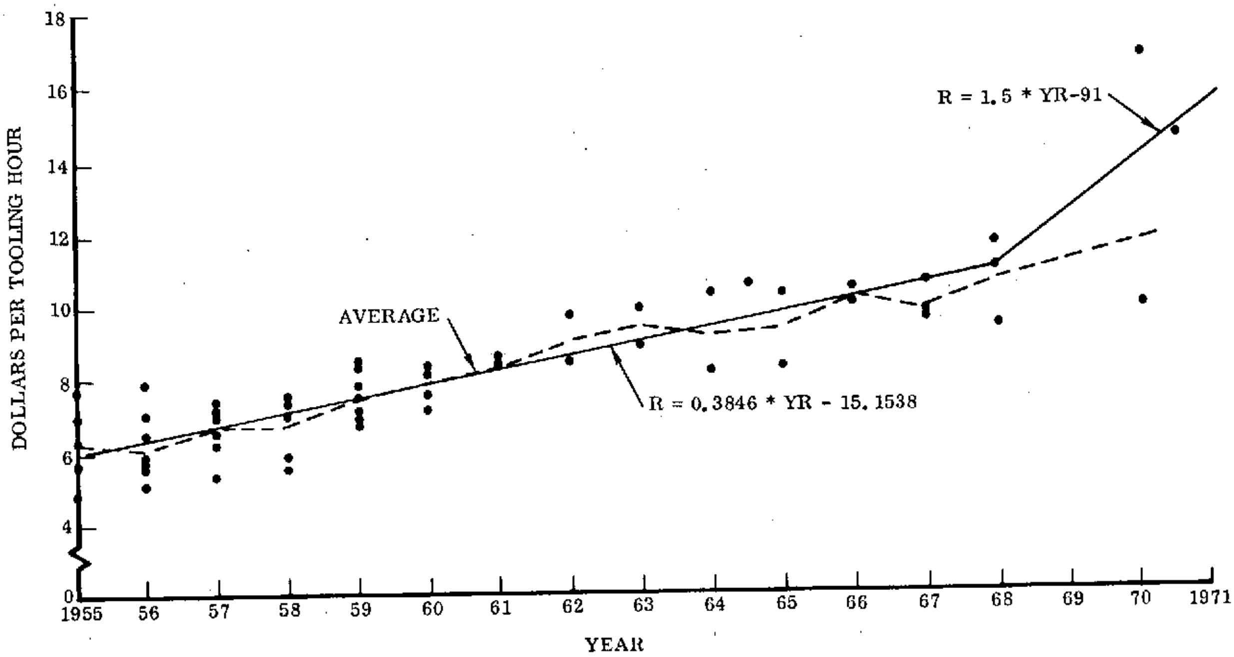


Figure 2-45. Tooling Labor Rate Versus Year

manufacturing costs (wing, body, horizontal, vertical, and nacelle), initial and sustaining engineering costs, basic tooling costs (basic tool engineering, manufacturing, and material) and rate and sustaining tooling costs. Table 2-27 summarizes the elements of the total vehicle program cost model.

As part of the total vehicle cost derivation a learning curve approach is applied to first unit costs to compute the cost of any subsequent unit or production lot. The learning curve analysis assumes a constant slope for the cumulative unit average cost (cumulative total cost divided by cumulative number of shipsets) plotted against shipset. This, in effect, assumes that a percentage increase in production results in a constant percentage decline in the average unit cost. The cumulative average cost for all units through the Nth unit then can be presented as a function of the first unit cost (FUC) and the learning curve slope (S) as follows:

cumulative average cost = FUC * N ** B

where

B = ALOG (S)/ALOG (2)

The corresponding total cost of N units is N times the cumulative average cost through Nth unit. The actual unit cost for the Nth unit is:

unit cost = FUC * (N ** (B+1) - (N-1) ** (B+1))

$\approx$ FUC * (B+1) * N ** B   for N > 16

Table 2-27. Total Vehicle Program Cost Model

NONRECURRING RDTE	RECURRING RDTE AND PRODUCTION (Contd)
Precontract Funded Studies	Hydraulics/Pneumatics
Airframe Development	Electrical/Electronics
Initial Engineering	Instruments
Development Support	Armament
Engineering Material	Engine Associated Equipment
Manufacturing Support and Material	Fuel System
Quality Control	Avionics Provisioning
Basic Tooling	Furnishings/Equipment
Basic Airframe Tool Manufacturing	Engine Production
Basic Subsystem Tool Manufacturing	Avionics Production
Basic Tool Engineering	Armament
Tooling Material	Primary and Final Assembly
Manufacturing Development	Mission Equipment Installation
Plant Engineering and Material	Acceptance Operations
Propulsion Development	Sustaining Engineering
Avionics Development	Rate Tooling
Systems Engineering and Management	Sustaining Tooling
AGE Development and Procurement	Spares for Test
Training Equipment Development and Procurement	AGE for Test
Flight Test Operations	Technical Data
Technical Data	Program Management
Total Nonrecurring RDTE Costs	Total Flyaway Costs
	Support Costs
RECURRING RDTE AND PRODUCTION	Initial Spares and Replenishment Parts
Production Airframe	Airframe
Basic Structure	Propulsion
Wing	Avionics
Body	AGE for Production
Horizontal	Training Equipment
Vertical	Test Aircraft Conversion
Nacelle	Category II and III Test Support
Subsystems	
Gear	Total Support Costs
Surface Controls	Total Program Costs
Environmental Systems	

Table 2-28. Total Vehicle Cost Elements Established by Direct Input

Precontract Funded Studies
Systems Engineering and Management
Training Equipment Development and Procurement
Avionics Production
Program Management
Category II and III Test Support

Total vehicle cost elements, which are input directly, are summarized in Table 2-28. Following are the expressions utilized to compute the remaining cost elements. Variable definitions are listed at the end of the section or with the dictionary of input parameters.

Engineering Material

FE2 * ENGRS

Manufacturing Support and Material

18.76 * ENGRHR * (1 + EIFAC) ** (YR-70)

Quality Control

(FQ1 * ENGRHR + FQ2 * TOOLHR) * MRATE

Basic Subsystem Tool Manufacturing

STF * CMT * WTSYS ** C

Manufacturing Development

FT4 * TOOLHR * MRATE

Plant Engineering and Material

FT3 * TOOLHR * (MRATE + 2.00)

Propulsion Development

.295E8 * (FP/1000) ** .55 * MACHNO ** .66 * SHP (6) * QENG * (1+SPRS) ** .1

Avionics Development

.55ES * WI ** .439 + .375E6 * WL ** .439

AGE Development and Procurement

.05 * ADDE + .15 * FV

Flight Test Operations

.75 * SHP (6) ** 1.1 * TAKOFF ** .08 * VELALT ** .9

Technical Data

.02 * FV

Subsystems First Unit Cost

C1XX * C2XX * P1XX * (YY/P1XX) ** C3XX

where	XX	=	LG	=	WG	(Landing Gear)
			SC		WS	(Surface Controls)
			AC		WO	(Environmental Systems)
			HP		WQ	(Hydraulics/Pneumatics)
			EE		WE	(Electrical/Electronics)
			IN		WI	(Instruments)

AR	WARMA	(Armament)
EN	WPSYS	(Engine Associated Equipment)
FS	WPFS	(Fuel System)
AV	WL	(Avionics Provisioning)
FE	WF	(Furnishings and Equipment)

Engine Production

3270 * CFENG * FP ** .60

Armament

(Assumed zero at present)

Primary and Final Assembly

FFA * (CFUAF + CFUSS)

Mission Equipment Installation

FAA * (CFUAV + CFUAR)

Acceptance Operations

FAO * (CFUAF + CFUAV + CFUENG + CFUINS + CFUSS + CFUSSY)

Spares for Test

(F3 * AFT + F4 * ENG $\pm$ AIS * AV)/(SHP(6)) ** .7

AGE for Test

P2 * AGTA * AFT + P4 * AGTP * ENG + P6 * AGTV * AV

Technical Data

F11 * ATOT

Airframe Spares and Replenishment Parts

PS * AFT

Propulsion Spares and Replenishment Parts

F6 * ENG

Avionics Spares and Replenishment Parts

F7 * AV

AGE for Production

F8 * (1 + F9) * ATOT

Training Equipment

F10 * ATOT

Test Aircraft Conversion

F12 * ATOT

where ADDE is the sum of the costs for precontract funded studies and initial engineering ($)

AFT is the airframe production cost ($)

ATOT is the sum of production costs for the airframe, avionics, and propulsion system ($)

AV is the avionics production costs ($)

CFUAF is the first unit cost of the airframe ($)

CFUAR is the first unit cost of the armament ($)

CFUAV is the first unit cost of the avionics ($)

CFUENG is the first unit cost of the propulsion system ($)

CFUINS is the cost of mission equipment installation ($)

CFUSS is the first unit cost of the subsystems ($)

EIFAC is the annual rate of inflation of engineering labor rates

ENG is the propulsion system production cost ($)

ENGRHR is the total number of initial engineering hours (HR)

ENGRS is the total initial engineering cost ($)

EQUIP is the total weight of the vehicle systems and equipment (lb)

FP is the total sea level thrust (lb)

FV is the total aircraft production cost ($)

MACHNO is the Mach number

MRATE is the quality control labor rate ($/HR)

QENG is the number of engines per aircraft

SHP(6) is the total number of shipsets to be produced

TAKOFF is the vehicle gross takeoff weight (lb)

TOOLHR is the sum of basic tool manufacturing hours for airframe and subsystems, and rate tooling hours (HR)

VELALT is the maximum velocity of cruise altitude (kn)

WARMA is the weight of the vehicle armament (lb)

WE is the weight of the vehicle electrical system (lb)

WF is the weight of the vehicle furnishings (lb)

WG is the weight of the vehicle landing gear (lb)

WI is the weight of the vehicle instruments (lb)

WL is the weight of the vehicle avionics (lb)

WO is the weight of the vehicle environmental system (lb)

WPFS is the weight of the vehicle fuel system (lb)

WPSYS is the weight of the vehicle propulsion system equipment (lb)

WQ is the weight of the vehicle hydraulics/pneumatics system (lb)

WS is the weight of the vehicle surface controls(lb)

WTSYS is the sum of the vehicle weights for subsystems (lb)

2.3.6 RETURN-ON-INVESTMENT ANALYSIS

2.3.6.1 Direct Operating Cost. The direct operating cost computation requires as input the aircraft price, as previously computed in the total vehicle cost module, and aircraft performance, defined principally as fuel and time required for various distance increments up to the operational range. The input when applied to the 1967 Air Transport Association formula (Reference 18) develops direct operating cost elements for specified distance increments. The Air Transport Association formula provides the basis to compute crew cost (primarily a function of the number in the cockpit crew), time to cover specified distances, and aircraft gross weight. Fuel and oil costs are computed directly from block fuel required. Insurance (hull insurance only, liability is an indirect cost) is computed as an annual percentage of the aircraft price. Maintenance is computed as a function of time, weight, thrust, and hardware cost. Depreciation is computed for a specified number of years, and includes depreciation of spares as well as primary flight equipment. The resultant output is direct operating cost per aircraft mile and per available seat block for various distances up to the operational range of the airplane.

2.3.6.2 Return-On-Investment. To compute return-on-investment data, a comparison is made between revenue and direct plus indirect operating costs. City pair traffic data, distances, and fare formula establish the revenue of interest. Aircraft capacity, frequency, and load factor constraints determine the required flight frequency, indirect costs, and fleet size. Indirect costs are generally computed in accordance with the Lockheed formula (Reference 19) by city pair for such factors as aircraft servicing, stewardess expense, food, reservations and sales, baggage handling, and general and adminstrative expenses.

To compute return-on-investment, total income minus total cost is compared to total investment as determined by fleet size, aircraft price, and spares factors. Return-on-investment is calculated as that percentage return on net invested capital (initial investment minus cash flow from depreciation) that would equal the same percentage return on fixed return investment, such as an accrual savings deposit. Return-on-investment is computed for each city pair and for the entire system. In this way, it is possible to establish the traffic and distance requirements to make a given aircraft profitable and to make a meaningful comparison between two airplanes where seating capacity, performance, and price are different.

SECTION 3

COMPUTER PROGRAM

The results of this study were programmed for a CDC 6000 series digital computer and a CDC 250 processor/CDC 252 CRT display console. A block diagram illustrating the program and the flow of information between subroutines is presented in Figure 3-1. Input is read utilizing the NAMELIST library subroutine. A total of 15 separate functional blocks of data are required. Output is comprised of the following:

Group Weight Statement Geometry Data Performance Data Balance Data	Vehicle Synthesis
Loads Data Geometry Data Theoretical Weight Data	Structural Synthesis
Theoretical Weight Data Actual Weight Data Material Purchase Weight Data Parts Listing	Parts Definition
Manufacturing Costs Material Costs Engineering Costs Tooling Costs Total Vehicle Program Costs Return-on-Investment	Cost Analysis

In formulating the logic processes and communication links used by the program, several ground rules were followed. It was the intention to make the program as flexible as possible from the user's (application) standpoint and also from the programmer's (modification, update) standpoint. The actual program deck is comprised of nearly 200 functionally independent subroutines. This high degree of modularity provides a means of program updating or modification simply by removing a complete subroutine and replacing it with a new version. The new subroutine, which is restricted only to retaining the same input/output communication links, may deffer by only a card or two, or may pursue a whole new analysis procedure.

A special feature of the program is its ability to generate much of its own required data. This is a result of the coupling of several levels of synthesis routines, each acting as the driver for the one that follows. The output from the vehicle synthesis is used as input by the structural synthesis routines, whose output in turn is used as input by the

parts definition routines. This procedure results in input requirements that are, for the most part, on a very generalized, descriptive level consistent with typical preliminary design data. Included as part of this capability is the concept of optional input. The optional portion of the input is comprised of a series of parameters that are not always known during initial vehicle studies. Capability is built into the program to automatically calculate typical values for these parameters if they are not input, but if they are available and input, the internal calculation is suppressed and the input value is utilized.

It was found that because of the degree of detail considered, the cumulative volume of output data available often became burdensome. A capability to suppress portions of the output was built into the program. This allows the user to tailor the output to the job at hand. Any combination of output may be selected, from a complete and fully detailed version, to a version consisting of a series of summary sheets.

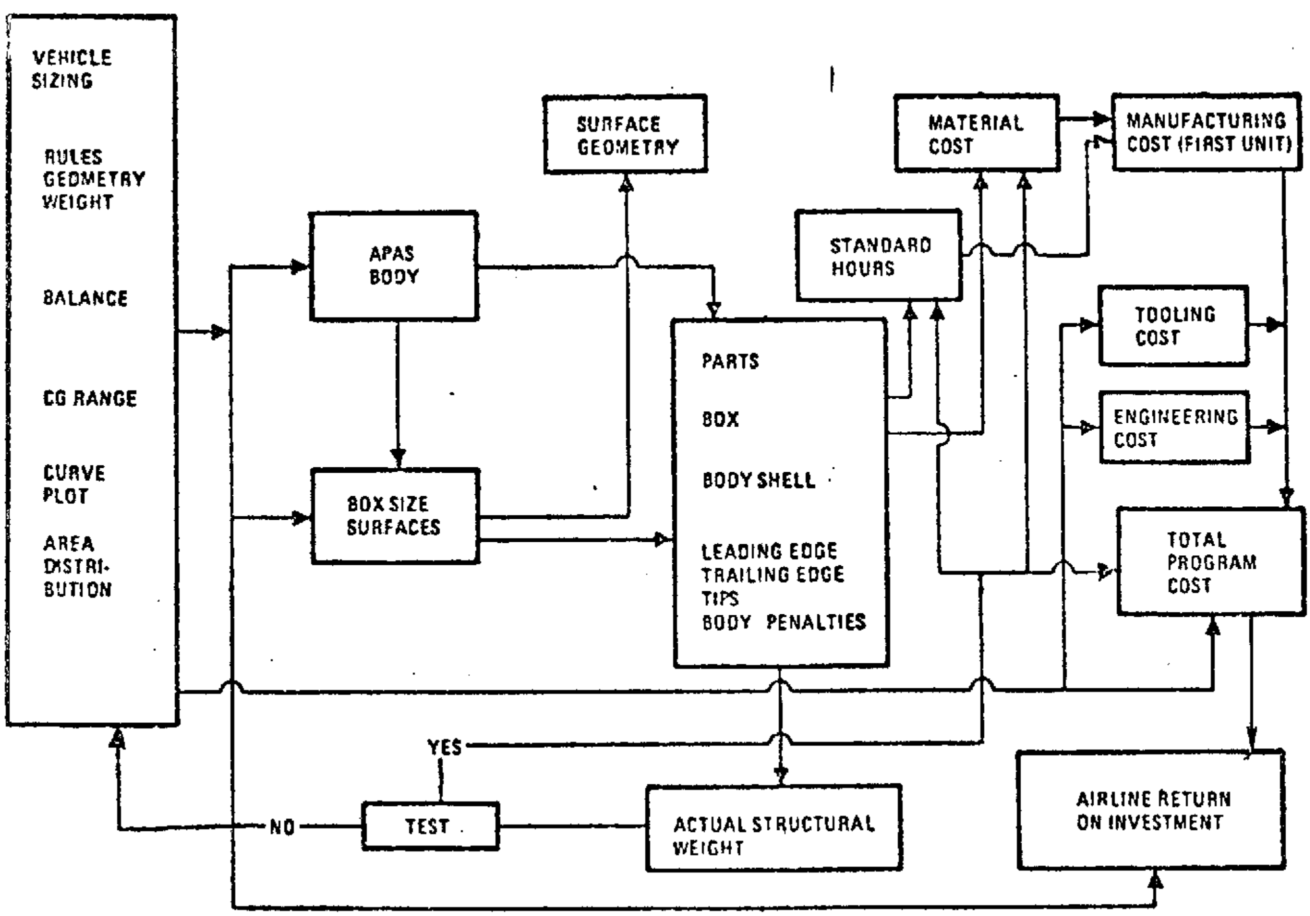


Figure 3-1. Information Flow Between Functional Blocks

3.1 OVERLAY STRUCTURE

The program was designed for use in an overlay mode and uses peripheral disk files for computed data storage. The primary overlay structure is illustrated in Figure 3-2. In this configuration the program will operate within a maximum core allowance of 60 thousand octal words.

The program driver, overlay 0, 0 is resident with all primary overlays and acts as a main control routine for the total program. It establishes a communication link, and provides a logic flow for the program from beginning to end. The functional logic associated with the program is derived from four basic modules. These modules include vehicle synthesis, structural synthesis, part definition and cost.

The vehicle synthesis program sizes the aircraft, performs a balance analysis, distributes the area, and displays a planform view along with pertinent design data, such as weight statements, balance data, and general geometric data. The program also includes a curve plotting routine that allows the user to perform parametric trade studies and obtain hard copy output for evaluation.

This program has been divided into two separate primary overlays. The 2,0 overlay includes all functions associated with vehicle sizing except area distribution. The area distribution subroutines are in 5, 0 overlay. The overlay structure for vehicle sizing is shown in Figure 3-3 (overlay 2, 0) and the area distribution overlay structure is shown in Figure 3-4 (overlay 5, 0).

The structural synthesis program provides detailed geometry, loads, and weight data for the primary structural elements associated with the aerodynamic surfaces and the basic fuselage structural shell. The structural synthesis provides a means of descriptively designing structural components that fulfill specified requirements of strength and geometry.

The structural synthesis process is comprised of two subprograms, one for the aerodynamic surface structural box and one for the basic fuselage structural shell. The subroutines for the aerodynamic surface structural box are in overlay 3,0 and the subroutines for the basic fuselage shell is in overlay 6,0. The overlay structure for structural synthesis is shown in Figure 3-5 (overlay 3,0) and Figure 3-6 (overlay 6,0).

The part definition program utilizes output from the structural synthesis to derive detail parts sufficient to construct the complete assembly. The actual weights and the weight of the raw material to be purchased is also derived as part of the part definition processes based on the computed part geometry.

The part definition program is coupled with the cost program through the manufacturing cost analysis. The manufacturing cost analysis consists of a definition of manfacturing processes associated with each part, the standard hours associated with the process and

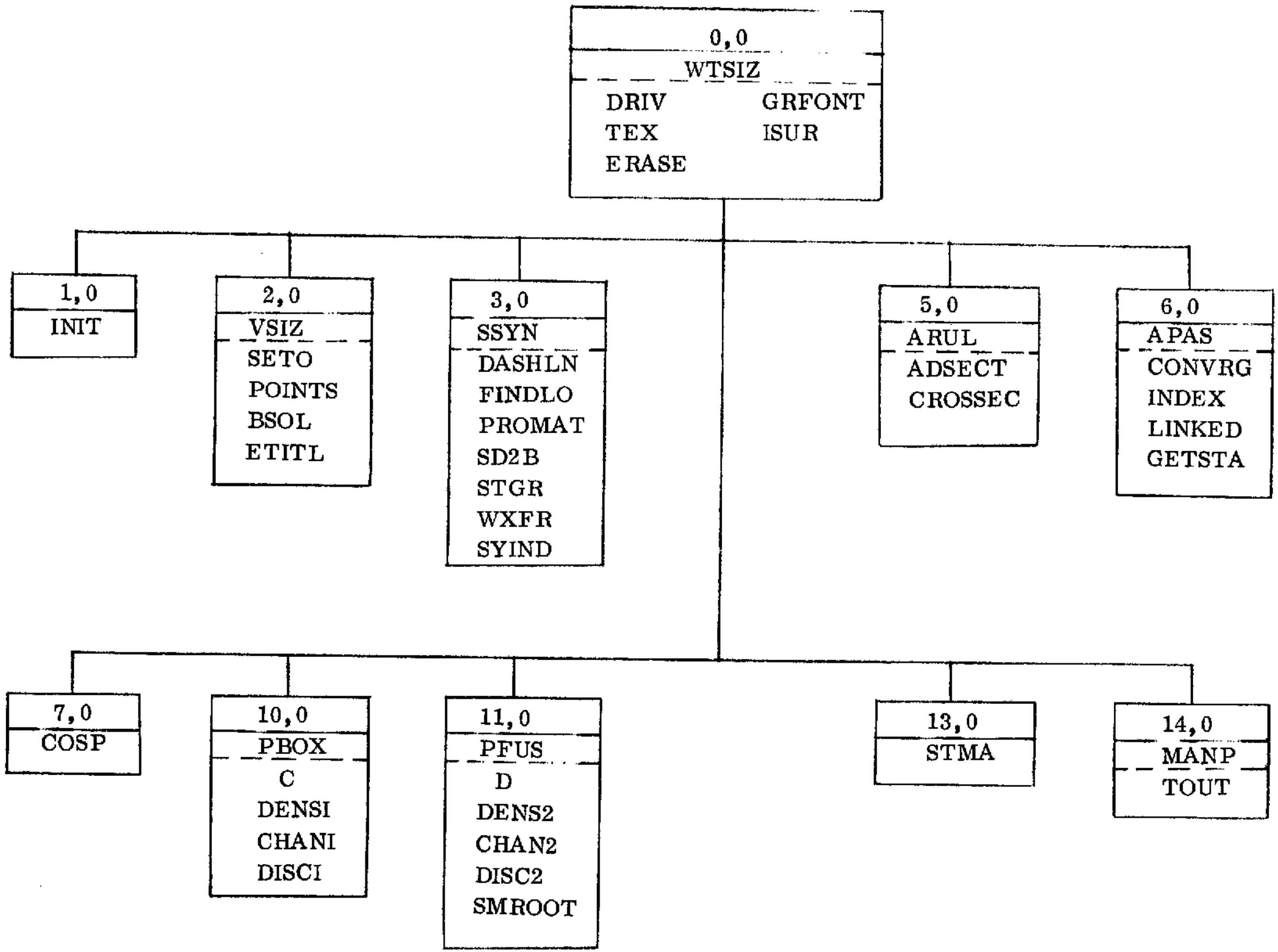


Figure 3-2 Primary Overlay Structures

the material weight. Additional cost analysis includes tooling, engineering, total program and return-on-investment.

The part definition/cost synthesis program is overlayed using five (5) primary overlays. These consist of the cost analysis (overlay 7, 0), the box structure part definition (overlay 10, 0), the fuselage part definition (overlay 11, 0), the manufacturing cost (overlay 13, 0) and the manufacturing processes (overlay 14, 0). The overlay structure for the parts definition/cost synthesis is shown in Figure 3-7 (overlay 7, 0) and Figure 3-8 (overlay's 10, 0; 11, 0; 13, 0; and 14, 0).

3.2 DESCRIPTION OF SUBROUTINES

The program was constructed in a highly modular format. It consists of a main program driver and nearly 200 subroutines, each made as independent as practical with respect to the total program. The overlay structure for each of the primary overlays is shown in Figures 3-3 through 3-8.

The program driver, overlay 0,0 is resident with all primary overlays and is used to call 1,0. The description of overlays 0,0 and 1,0 is as follows:

OVERLAY 0,0 (PROGRAM WTSIZ)

This is the primary overlay for the entire graphics program. There are five subroutines resident within this overlay that are essential to each operation within the overall program. The resident subroutines are DRIV, TEX, ERASE, GRFONT and ISUR.

The subroutine DRIV displays the four options available within the program. These options are the vehicle sizing portion (Overlay's 2,0-VSIZ and 5,0-ARUL), the aerodynamic surface structural synthesis portion (Overlay 3,0-SSYN), the fuselage structural synthesis portion (Overlay 6,0-APAS), and the cost portion (Overlay 7,0-COSP, 10,0-PBOX, 11,0-PFUS, 13,0-STMA, and 14,0-MAND). The main menu for each of the options is also called from this subroutine.

The subroutines TEX, ERASE, and GRFONT are for use with interactive graphics. TEX is used for displaying the various routines, ERASE causes the erasures of each display, and GRFONT produces the picking font utilized to implement changes.

The subroutine ISUR is utilized as a mass storage communication link.

OVERLAY 1,0 (INIT)

This overlay is called from DRIV and is used to initialize certain variables.

The description of the subroutine for the vehicle synthesis, structural synthesis and detail parts/cost are divided into the following categories:

Vehicle Synthesis	-	Overlays 2,0 and 5,0
Structural Synthesis	-	Overlays 3,0 and 6,0
Part Definition/Cost	-	Overlays 7,0; 10,0; 13,0; 14.0

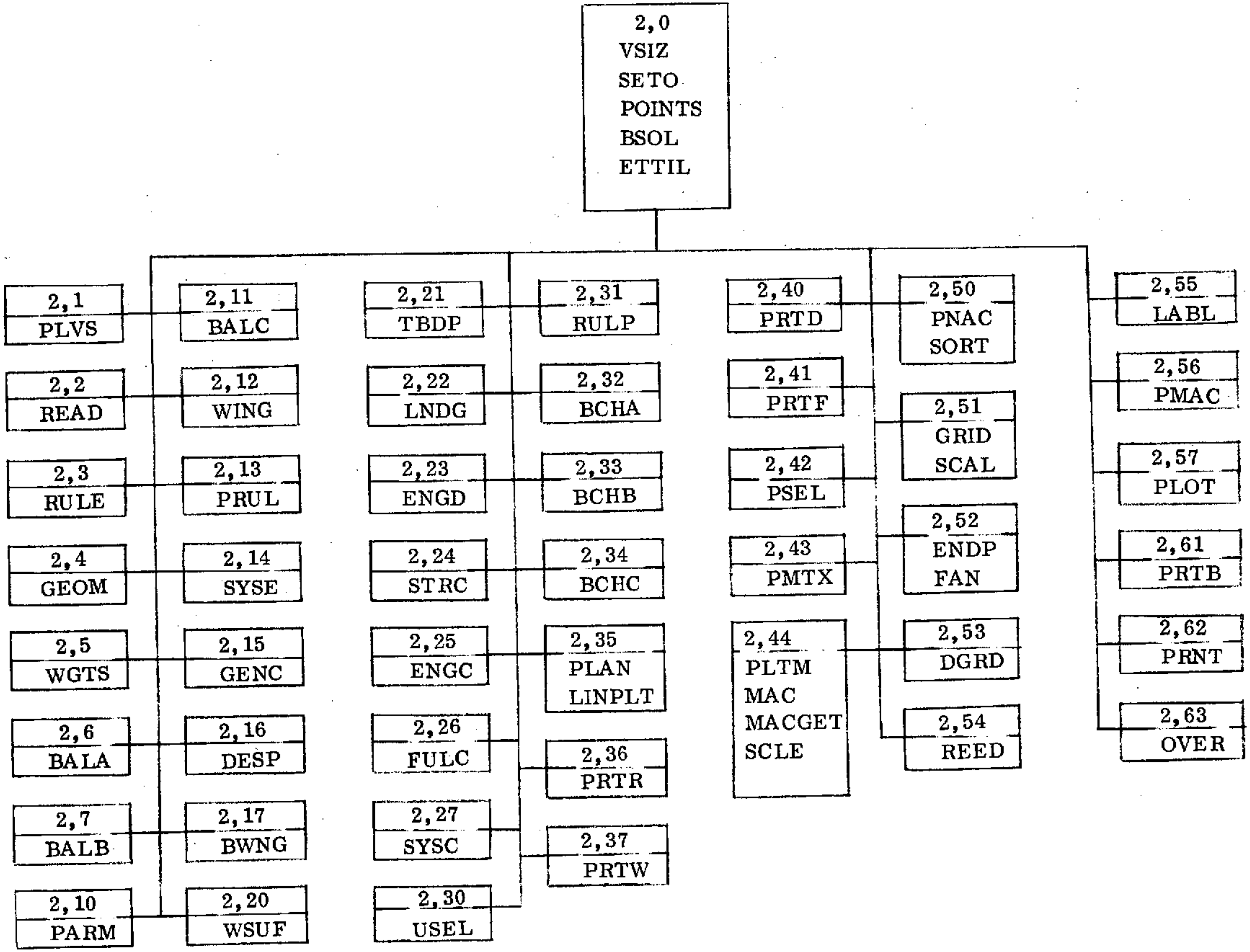


Figure 3-3 Overlay Structure for Vehicle Sizing

OVERLAY 2,0 (VSIZ)

This subroutine functions as the main display routine for the vehicle sizing program. Entry to all other subprograms generally originates from here; to proceed from subprogram to subprogram, a return must first be made to this program in order to call the next routine desired. The iterative process is accomplished through this routine. This overlay also includes subroutine SETO (sets core to zero), POINTS (lease square curve fit routine), BSOL (does general calculations for balance programs), and ETITL (sequence titles on selected plotting axis).

OVERLAY 2,1 (PLVS)

This subroutine indexes input variables for the selection of various plotting sequences.

OVERLAY 2,2 (READ)

This subroutine reads the initial input parameters using the namelist library routine. The input parameters were described and their setup is discussed in Section 3.

OVERLAY 2,3 (RULE)

This subroutine provides initial sizing approximations of performance, propulsion, and loads.

OVERLAY 2,4 (GEOM)

This subroutine provides the necessary geometry to the weight routine and sufficient data to display a three-view drawing of the sized aircraft. It containes the geometry equations as discussed in Section 2.1 of Volume II.

OVERLAY 2,5 (WGTS)

This subroutine contains the weight equations as discussed in Section 2.2 of Volume II. It provides weight data for the summary and detail weight statements according to MIL-STD-254.

OVERLAY 2,6 (BALA)

This subroutine contains the necessary equations to calculate the individual and composite C.G. locations for weight items classified under Wing and Contents. Subroutine ASOL, resident with BALA, performs general balance calculations for use with a detailed breakdown of the wing.

OVERLAY 2, 7 (BALB)

This subroutine contains the necessary equations to calculate the individual and composite C. G. locations for weight items classified under Body and Contents.

OVERLAY 2, 10 (PARM)

This subroutine displays the detail weight statement according to MIL-STD-254 for the entire aircraft at various weight conditions, and it also displays the C. G. locations of each item contained in the weight statement.

OVERLAY 2, 11 (BALC)

This subroutine contains the necessary equations to calculate the aircraft C. G. location for weight empty, basic operating weight, zero fuel weight, and mission takeoff weight conditions.

OVERLAY 2, 12 (WING)

This subroutine displays the detail weight statement according to MIL-STD-254 for the wing and detailed dimensional data for the aircraft.

OVERLAY 2, 13 (PRUL)

This subroutine displays the detail weight statement according to MIL-STD-254 for the propulsion system and detailed dimensional data for the aircraft.

OVERLAY 2, 14 (SYSE)

This subroutine displays the detail weight statement according to MIL-STD-254 for the remaining systems and equipment that comprise the weight empty condition of the aircraft and detailed dimensional data for the aircraft.

OVERLAY 2, 15 (GENC)

This subroutine displays general aircraft parameters from the input list. These parameters would most likely be involved with aircraft tradeoff studies. All values displayed can be changed and resizing initiated.

OVERLAY 2, 16 (DESP)

This subroutine displays the design parameters from the input list. All values displayed can be changed and resizing initiated.

OVERLAY 2, 17 (BWNG)

This subroutine displays the basic wing parameters from the input list. All values displayed can be changed and resizing initiated.

OVERLAY 2, 20 (WSUF)

This subroutine displays the wing surfaces parameters from the input list. All values displayed can be changed and resizing initiated.

OVERLAY 2, 21 (TBDP)

This subroutine displays the tail and body parameters from the input list. All values displayed can be changed and resizing initiated.

OVERLAY 2, 22 (LNDG)

This subroutine displays the landing gear parameters from the input list. All values displayed can be changed and resizing initiated.

OVERLAY 2, 23 (Subroutine ENGD)

This subroutine displays the engine data parameters from the input list. All values displayed can be changed and resizing initiated.

OVERLAY 2, 24 (Subroutine STRC)

This subroutine displays the structural coefficient parameters from the input list. All values displayed can be changed and resizing initiated.

OVERLAY 2, 25 (Subroutine ENGC)

This subroutine displays the engine coefficient parameters from the input list. All values displayed can be changed and resizing initiated.

OVERLAY 2, 26 (Subroutine FULC)

This subroutine displays the fuel system coefficient parameters from the input list. All values displayed can be changed and resizing initiated.

OVERLAY 2, 27 (Subroutine SYSC)

This subroutine displays the system coefficient parameters from the input list. All values displayed can be changed and resizing initiated.

OVERLAY 2, 30 (Subroutine USEL)

This subroutine displays the useful load parameters from the input list. All values displayed can be changed and resizing initiated.

OVERLAY 2, 31 (Subroutine RULP)

This subroutine displays the rules parameters from the input list. All values displayed can be changed and resizing initiated.

OVERLAY 2, 32 (BCHA)

This subroutine displays the balance parameters classified under Wing and Contents - Weight Empty from the input list. All values displayed can be changed and rebalancing initiated.

OVERLAY 2, 33 (BCHB)

This subroutine displays the balance parameters classified under Body and Contents - Weight Empty from the input list. All values displayed can be changed and rebalancing initiated.

OVERLAY 2, 34 (BCHC)

This subroutine displays the balance parameters for basic operating weight, zero fuel weight, and mission takeoff weight conditions of the aircraft from the input list. All values displayed can be changed and rebalancing initiated.

OVERLAY 2, 35 (PLAN)

This subroutine displays a planform of the aircraft (only after the aircraft has been balanced) from the data received from the geometry and balance routines. It also displays certain geometry information related to the planform, as well as a summary of the aircraft's balance data at each weight condition of the aircraft. The plotting arrangement is controlled within this subroutine by subroutine LINPLT, which is resident.

OVERLAY 2, 36 (PRTR)

This subroutine is a print routine that works along with PRTW, PRTD, PRTF, and PRTB when PRINT is called from the main menu. It specifically controls the printing of that data associated with the rules routine.

OVERLAY 2, 37 (PRTW)

This subroutine is a print routine that works along with PRTR, PRTD, PRTF, and PRTB when PRINT is called from the main menu. It specifically controls the printing of the group weight statement and its corresponding data.

OVERLAY 2, 40 (PRTD)

This subroutine is a print routine that works along with PRTR, PRTW, PRTF, and PRTB when PRINT is called from the main menu. It specifically controls the printing of all dimensional data associated with the overall aircraft.

OVERLAY 2, 41 (PRTF)

This subroutine is a print routine that works along with PRTR, PRTW, PRTD, and PRTB when PRINT is called from the main menu. It specifically controls the printing of additional detailed aircraft data.

OVERLAY 2,42 (PSEL)

This is an interactive graphics display routine that displays the menu for the plot selections.

OVERLAY 2,43 (PMTX)

This is an interactive graphics display routine that displays the plotting matrix and the calculated values for the curves.

OVERLAY 2,44 (PLTM)

This is the interactive graphics plotting routine that sets up the grid and plots the curves. This overlay also includes subroutine MAC (defines and stores the plot grid and scale tic marks), MACGET (retrieves data from subroutine MAC), and SCLE (defines the plot scales increments for the position of the lines displayed between the maximum and minimum grid values.

OVERLAY 2,50 (PANC)

This subroutine reads in, and along with its subroutine SORT, sorts the data used in the displaying of the C.G. range curve into most forward to most aft position.

OVERLAY 2,51 (GRID)

This subroutine calculates the position of the grid lines for the C.G. range plot, and along with its subroutine SCAL, provides even plot scale increments for the position of the lines displayed between the maximum and minimum grid values.

OVERLAY 2,52 (ENDP)

This subroutine, along with its subroutine FAN, computes the maximum and minimum values of the grid lines to be displayed on the C. G. range diagram from the input data read in.

OVERLAY 2,53 DGRD)

This subroutine displays the grid based on the values and increments calculated in GRID and ENDP.

OVERLAY 2,54 (REED)

This subroutine prepares the C. G. range input data to be displayed in the sorted order achieved in PANC.

OVERLAY 2,55 (LABL)

This subroutine numerically labels all horizontal and vertical C. G. grid lines.

OVERLAY 2,56 (PMAC)

This subroutine displays weights, C. G. locations, and names of all C. G. range items. All of the weights or C. G. locations may be changed and the grid rescaled and the data resorted prior to plotting.

OVERLAY 2,57 (PLOT)

This subroutine plots the data input for the various C. G. range items in the order that they have been sorted.

OVERLAY 2,62 (PRNT)

This subroutine is a print routine that prints all data associated with the C. G. range diagram when PRINT is called from that display.

OVERLAY 2,63 (OVER)

This subroutine causes two messages to appear before the user: 1) RESET, and 2) OVER. If RESET is picked, it resets the entire program back to the original input data state. This allows the user to perform a completely different tradeoff study utilizing the initial input data plus his new changes, without having to reset each parameter individually back to its initial state. If OVER is again picked, the console will release the program from the graphics terminal, as the user has indicated completion of the task.

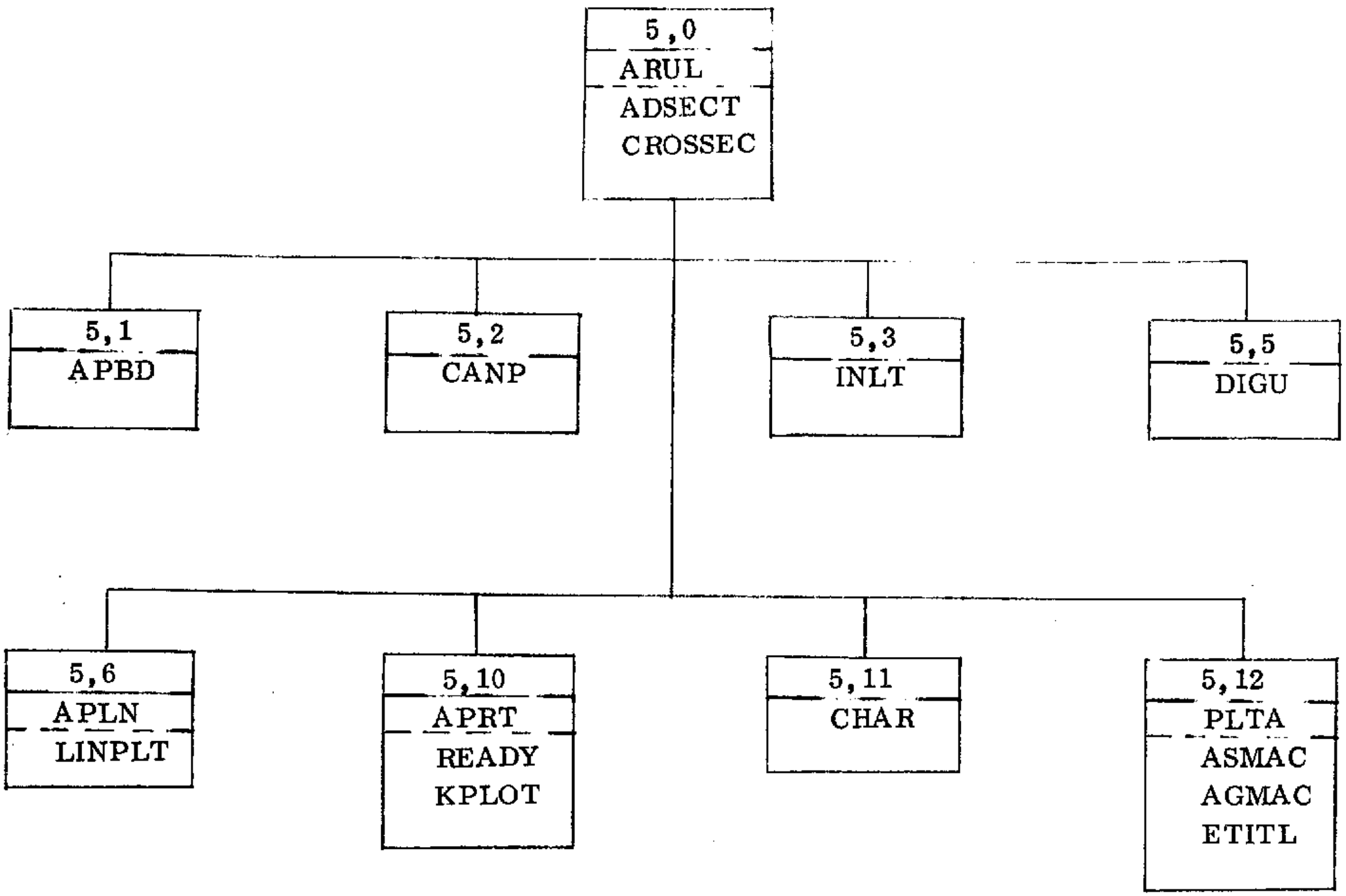


Figure 3-4 Overlay Structure for Area Distribution

OVERLAY 5,0 (ARUL)

This subroutine functions as the main driver for the area distribution program. Entry to all other subprograms associated with area distribution originate from here. All vehicle sizing data which supplements area distribution is conveyed from Overlay 2,0 (VSIZ) to this subroutine. This overlay also includes subroutine ADSECT (adds up cross sectional area data) and CROSSEC (generates the cross sectional area distribution for all airfoil surfaces).

OVERLAY 5,1 (APDB)

This subroutine computes the cross sectional area distribution for a fighter or transport body.

OVERLAY 5,2 (CANP)

This subroutine generates the cross sectional area distribution for a canopy.

OVERLAY 5,3 (INLT)

This subroutine generates the cross sectional area distribution for an inlet. This program is not used at the present time, but is provided, so the user can input an existing program which he possesses which does describe an inlet.

OVERLAY 5,5 (DIGU)

This subroutine transforms the radii array into inches (Langley) or DGU's (WPAFB) so that the body can be displayed.

OVERLAY 5,6 (APLN)

This subroutine displays a planform of the aircraft after the vehicle has been sized, balanced and area distributed. This overlay also includes subroutine LINPLT (controls plotting arrangement).

OVERLAY 5,10 (APRT)

This subroutine prints out the results of the area distribution on the body. This overlay also includes subroutine READY (prepares area distribution data for the printed output plot routine) and KPLOT (plots cross sectional area distribution of the aircraft for the printed output).

OVERLAY 5,11 (CHAR)

This subroutine displays the change parameters for the area distribution program. All values displayed can be changed and redistribution of the area initiated.

OVERLAY 5,12 (PLTA)

This subroutine controls the plotting of the area distribution curves on the graphics screen. It sets up the plot grid and actually plots the curves. This overlay also includes subroutines ASMAC (stores data defining grid hash marks), AGMAC (retrives data defining grid has marks) and ETITL (sequences titles on selcted plotting axis).

OVERLAY 3,0 (SSYN)

This subroutine is the main display driver that provides selection communication with the various structural synthesis math models and displays utilized within the aerodynamic surface structural synthesis. Subroutines resident within this overlay includes DASHLN (graphics program to display dashed lines), FINDLO (provides net shear, moment and torque loads information), PROMAT (manipulates selected material properties that are stored as a function of temperature), SD2B (transfers structural synthesis data to and from the holding buffer), STGR (evaluates available stringer types when skin-stringer construction type is used), WXFR (transfers structural synthesis data from the mass storage buffer to core), and SYINP (sets up aerodynamic surface structural synthesis input data, resets input data that has been changed on the screen to the original values, or zero's out common data).

OVERLAY 3,1 (MTLD)

This subroutine is a library of material properties.

OVERLAY 3,2 (GEOW)

This subroutine converts geometric parameters necessary to describe an aerodynamic surface to an elastic axis reference system.

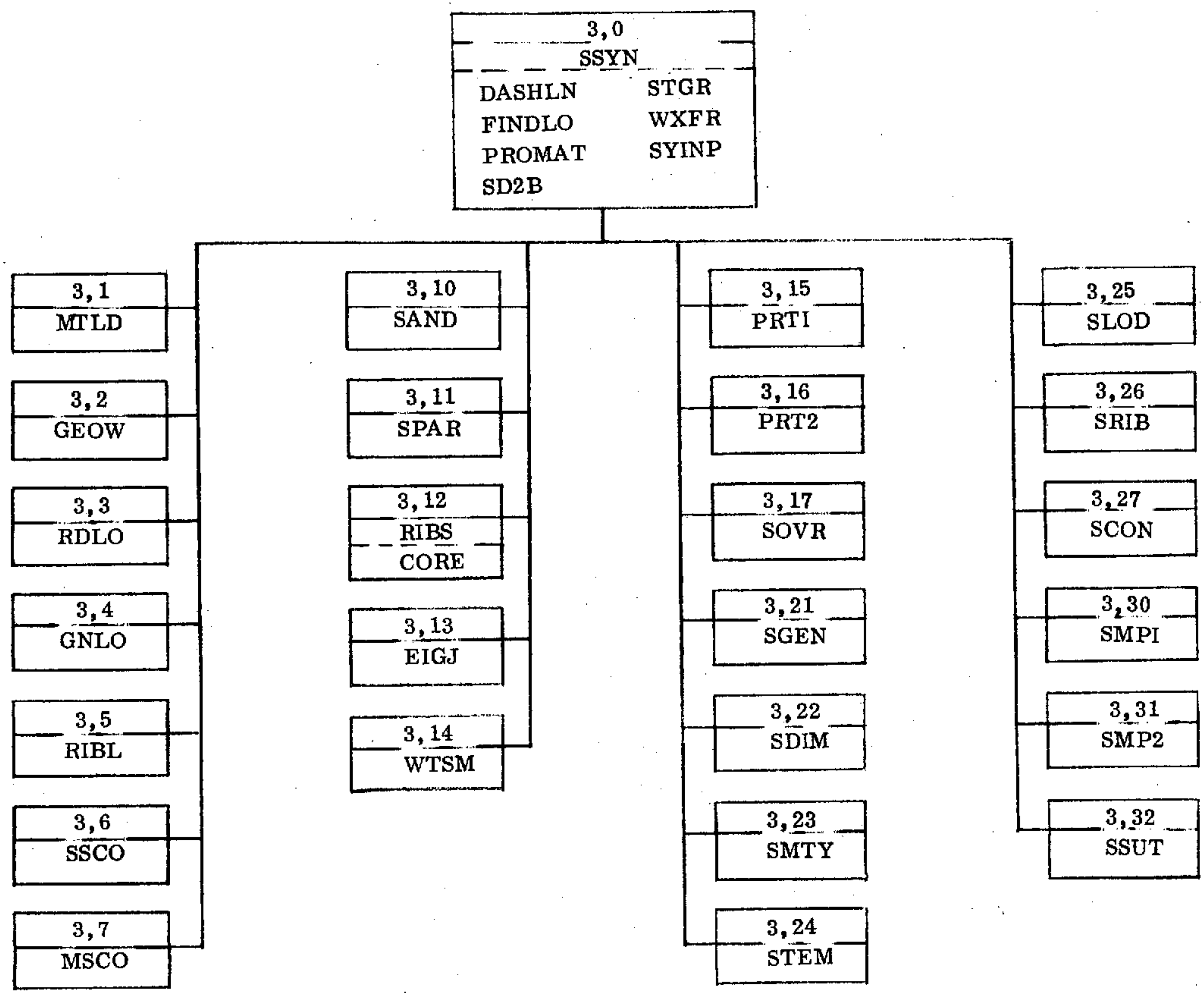


Figure 3-5 Overlay Structure for Aerodynamic Surface Structural Analysis

OVERLAY 3,3 (RDLO)

This subroutine is used to develop load envelopes by comparing given load conditions when the shear, moment, and torque load curves are input at specified stations.

OVERLAY 3,4 (GNLO)

This subroutine generates shear, moment and torque values at 10% increments from total surface loads, concentrated loads, inertia relief due to fuel and structure, and sustained load factor.

OVERLAY 3,5 (RIBL)

This subroutine determines spanwise locations of ribs as a function of given rib locations, or given rib pitch, or calculates rib pitch.

OVERLAY 3,6 (SSCO)

This subroutine sizes the skin-stringer covers. It utilizes the stringer properties read in from STGR (resident in Overlay 3,0) and sizes them to support the compressive load covers for the required rib pitch. Tension covers are sized for maximum tension or reverse bending. Tail surfaces utilize symmetrical loading conditions and symmetrical covers.

OVERLAY 3,7 (MSCO)

This subroutine sizes and weighs the upper and lower covers for multi-spar construction.

OVERLAY 3,10 (SAND)

This subroutine sizes and weights cover panels at 10% span locations for full depth sandwich construction.

OVERLAY 3,11 (SPAR)

This subroutine sizes and weights spar caps and webs utilizing internal load distribution as a function of construction mode. The bending material is distributed in the skin thus making spar caps sensitive to shear and minimum area requirements.

OVERLAY 3, 12 (RIBS)

This subroutine sizes and weights rib caps and webs. When the construction is full depth sandwich, subroutine CORE is utilized which is resident within this routine. Subroutine CORE selects honeycomb core density and computes the weight. Data tables are stored for aluminum and stainless steel honeycomb core mechanical properties as a function of temperature.

OVERLAY 3, 13 (EIGJ)

This subroutine determines stiffness parameters, bending (EI) and torsional (Gct), as a function of station location.

OVERLAY 3, 14 (WTSM)

This subroutine provides a summary of weight data from the structural synthesis computations.

OVERLAY 3, 15 (PRT 1) and 3, 16 (PRT 2)

The aerodynamic surface structural synthesis printout is divided into two parts. The first part is printed from overlay 3, 15 (PRT 1) and the second part from 3, 16 (PRT 2).

OVERLAY 3, 17 (SOVR)

This subroutine is an interactive graphics routine that provides the user a choice of either terminating the graphics job or resetting the current data back to the original values.

OVERLAY 3, 21 (SGEN)

This subroutine is an interactive graphics routine to display input and provide the user with the capability to select or change construction models.

OVERLAY 3, 22 (SDIM)

This subroutine is an interactive graphics routine to display input and provide the user with the capability to select or change dimensional data.

OVERLAY 3,23 (SMTY)

This subroutine is an interactive graphics routine to display input and provide the user with the capability to select or change construction material type.

OVERLAY 3,24 (STEM)

This subroutine is an interactive graphics routine to display input and provide the user with the capability to select or change material temperature selections.

OVERLAY 3,25 (SLOD)

This subroutine is an interactive graphics routine to display input and provide the user with the capability to select or change input loads data.

OVERLAY 3,26 (SRIB)

This subroutine is an interactive graphics routine to display input and provide the user with the capability to select or change rib stations.

OVERLAY 3, 27 (SCON)

This subroutine is an interactive graphics routine to display input and provide the user with the capability to select or change concentrated loads data.

OVERLAY 3,30 (SMPI) AND OVERLAY 3,31 (SMP2)

The material properties of selected material types are divided into two interactive graphics displays. The first half is displayed by overlay 3,30 (SIM1) and the second half by overlay 3,31 (SMP2). Both displays provide the user with the capability to select or change material property data.

OVERLAY 3,32 (SSUT)

This subroutine is an interactive graphics routine to display the output data for wing, horizontal, and vertical cover panels.

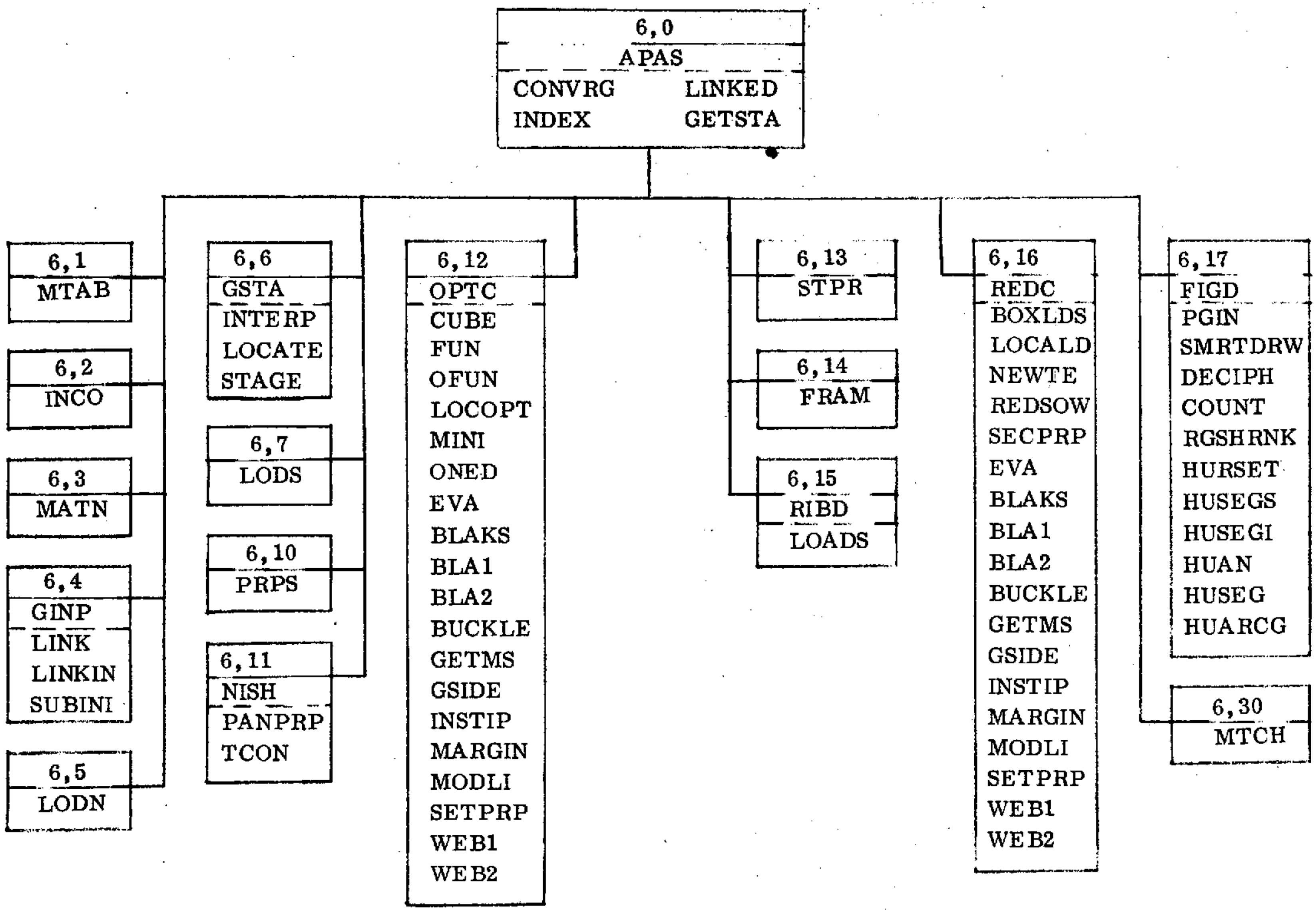


Figure 3-6 Overlay Structure for Fuselage Structural Synthesis

OVERLAY 6, 0 (APAS)

This subroutine is the overall driver program that calls the input, output and processing routines for the fuselage structural analysis. Subroutines resident within this overlay includes CONVRG (determines when to stop the redesign iteration process based on the input convergence criteria parameters), INDEX (determines the indices of the active design variables for a structural element), LINKED (finds all of the members of a given symmetry group), and GETSTA (finds the next station to be optimized based on input information).

OVERLAY 6,1 (MTAB)

This subroutine stores the library of material properties.

OVERLAY 6,2 (INCO)

This is an input control subroutine. This subroutine calls routines as required to read input data.

OVERLAY 6,3 (MATN)

This routine reads the material property input data.

OVERLAY 6,4 (GINP)

This subroutine is called by the input control subroutine INCON. It is used to read in the fuselage or aerodynamic surface basic geometry. Subroutines resident within this overlay includes LINK (calls an input routine which reads the information needed to set up the element symmetry groups. It then creates the symmetry indice arrays), LINKIN (reads symmetry group information used by subroutine LINK), and SUBINI (this routine is called by the geometry input subroutine GINPTI to read symmetry group input data).

OVERLAY 6,5 (LODN)

This subroutine is a general purpose external load input routine. It reads shear, bending moment, and torsion curves for the total structure.

OVERLAY 6,6 (GSTA)

This subroutine computes specific station geometry. Subroutines resident within this overlay includes INTERP (performs linear interpolations), LOCATE (locates the position of a given station within the stored geometry data array. It then determines the

required interpolation parameters needed to extract the station geometry information and STAGE (sets up station geometry when a new station is to be sized).

OVERLAY 6,7 (LODS)

This subroutine determines the externally applied loads at a given station by linear interpolation of input loads.

OVERLAY 6, 10 (PRPS)

This subroutine defines the material properties used during the analysis for each of the applied loading conditions at the temperature indicated for the loading condition.

OVERLAY 6,11 (NISH)

This subroutine initializes the variables used by the internal loads analysis routines. Subroutines resident within this overlay includes PANPRP (computes the material properties of a layered composite laminate) and TCON (initializes the geometry variables used by the analysis routines).

OVERLAY 6, 12 (OPTC)

This subroutine is the optimization control routine. It does an analysis of one structural element or symmetry group at a time until all elements at a given cross-section are optimized. Subroutines resident within this overlay includes CUBE, FUN, OFUN, LOCOPT, MINI, ONED, EVA, BLAKS, BLAI, BLAZ, BUCKLE, GETMS, GSIDE, INSTIP, MARGIN, MODLI, SETPRP, WEBI, and WEBZ. A brief description of these subroutines is as follows:

CUBE

This subroutine fits a cubic polynomial to four points and finds the minimum. The minimum found is compared with the minimum found on the previous call. If they agree within tolerance then the convergence flag is set. This routine is used by the one dimension search subroutine ONED.

FUN

This subroutine produces the gradient of the objective function being minimized at a given design point for use by the nonlinear math programming subroutine MINI.

OFUN

This subroutine is called from ONED during the one dimensional search. It calls FUN to evaluate various designs.

LOCOPT

This subroutine controls the local design of each symmetry group. It sets up the input required by the math programming subroutines and interprets the results.

MINI

The subroutine modifies the structural design of an element to maximize the margin of safety. The method of Davidon-Fletcher-Powell is used.

ONED

This subroutine is called from MINI. It obtains the interval in which a minimum lies and performs a one dimensional minimization.

EVA

This subroutine evaluates the overall acceptability of a structural element based on manufacturing constraints and stress analysis.

BLAKS

This subroutine stores the local buckling and crippling coefficients for the stiffened panel configurations.

BLA 1

This subroutine computes the section properties of the stiffened panel configurations.

BLA 2

This subroutine computes the critical local and general instability buckling stresses and the crippling stress for stiffened panel configurations.

BUCKLE

This subroutine performs a panel buckling analysis for general instability of simply supported curved orthotropic panels in bi-axial loading with shear.

GETMS

This subroutine computes the margins of safety for static loads for stiffened panels.

GSIDE

This subroutine determines the constraint function for manufacturing constraints such as minimum gage and maximum stiffener height etc.

INSTIP

This subroutine converts a set of design variables into a set of detail geometry dimensions. This routine acts as an interpreter between the math programming routine and the structural analysis routine.

MARGIN

This subroutine calculates the margins of safety of composite panels using an ultimate fiber strain criteria.

MODLI

This subroutine performs the stress analysis of spar caps and longerons.

SETPRP

This subroutine transfers the material properties of advanced composite materials into the local analysis variables.

WEB1

This subroutine translates the optimization variables into detail geometry variables and visa-versa. It is used for internal web elements.

WEB2

This subroutine performs stress analysis on internal shear web elements.

OVERLAY 6,13 (STPR)

This subroutine prints results at the end of each station optimization.

OVERLAY 6,14 (FRAM)

This subroutine sizes a frame based on shanley criteria and minimum gage constraints.

OVERLAY 6,15 (RIBD)

This subroutine synthesizes an aerodynamic surface rib. Also included within this overlay is subroutine LOADS which determines the externally applied loads at a given station by linear interpolation of input loads.

OVERLAY 6,16 (REDC)

This subroutine controls the resizing process. It calls the subroutines which add or subtract material from the structural elements in an attempt to produce a minimum weight structure. Subroutines resident within this overlay includes BOXLDS, LOCALD, NEWTE, REDSON, SECPRP, EVA. PLAKS, BLA1, BLA2, BUCKLE, GETMS, GSIDE, INSTIP, MARGIN, MODLI, SETPRP, WEB1, and WEB2. A brief description of these subroutines is as follows:

BOXLDS

This subroutine performs a box beam internal load solution at a cross-section and computes the complex bending stresses and shear flows for unit load components.

LOCALD

This subroutine calculates the load intensities and shear flows applied to the structural elements. These applied internal loads are based on the results of subroutine BOXLDS and the applied external loads from subroutine loads.

NEWTE

This subroutine predicts the cross-sectional area of a structural element necessary to produce a zero margin of safety.

REDSON

This subroutine is called by REDCON, it directs the resizing iteration process.

SECPRP

The subroutine computes the section properties for a one, two or three cell box beam.

EVA

This subroutine evaluates the overall acceptability of structural element based on manufacturing constraints and stress analysis.

BLAKS

This subroutine stores the local buckling and crippling coefficients for the stiffened panel configurations.

BLA 1

This subroutine computes the section properties of the stiffened panel configurations.

BLA 2

This subroutine computes the critical local and general instability buckling stresses and the crippling stress for stiffened panel configurations.

BUCKLE

This subroutine performs a panel buckling analysis for general instability of simply supported curved orthotropic panels in bi-axial loading with shear.

GETMS

This subroutine computes the margins of safety for static loads for stiffened panels.

GSIDE

This subroutine determines the constraint function for manufacturing constraints such as minimum gage, maximum stiffener height, etc.

INSTIP

This subroutine converts a set of design variables into a set of detail geometry dimensions. This routine acts as in interpreter between the math programming routine and the structural analysis routine.

MARGIN

This subroutine calculates the margins of safety of composite panels using an ultimate fiber strain criteria.

MODLI

This subroutine performs the stress analysis of spar caps and longerons.

SETPRP

This subroutine transfers the material properties of advanced composite materials into the local analysis variables.

WEB1

This subroutine translates the optimization variables into detail geometry variables and visa-versa. It is used for internal web elements.

WEB 2

This subroutine performs stress analysis on internal shear web elements.

OVERLAY 6,17 (FIGD)

The subroutine FIGD stores and retrieves the necessary instructions for drawing the structural element figures developed in the APAS structural analysis. Subroutines resident within this overlay includes PGIN (display driver for figure displays and parameter changes), SMRTDRW (driver for a series of routines used for displaying structural elements. All information is contained in the data base and is accessed as required), DECIPH (decodes information contained in the database), FGSHRNK (applies a user factor to shrink the size of the displayed figures) COUNT (used with subroutine DECIPH to keep track of where pointer is at during the decoding of the SMRTDRW drawing instructions). This overlay also includes several standard graphics routines. These standard routines are called by HURSET, HUSEGS, HUSEGI, HUAN, HUSEG, and HUARCG.

OVERLAY 6,30 (MTCH)

This is a graphics subroutine utilized to make material changes for use in the structural analysis.

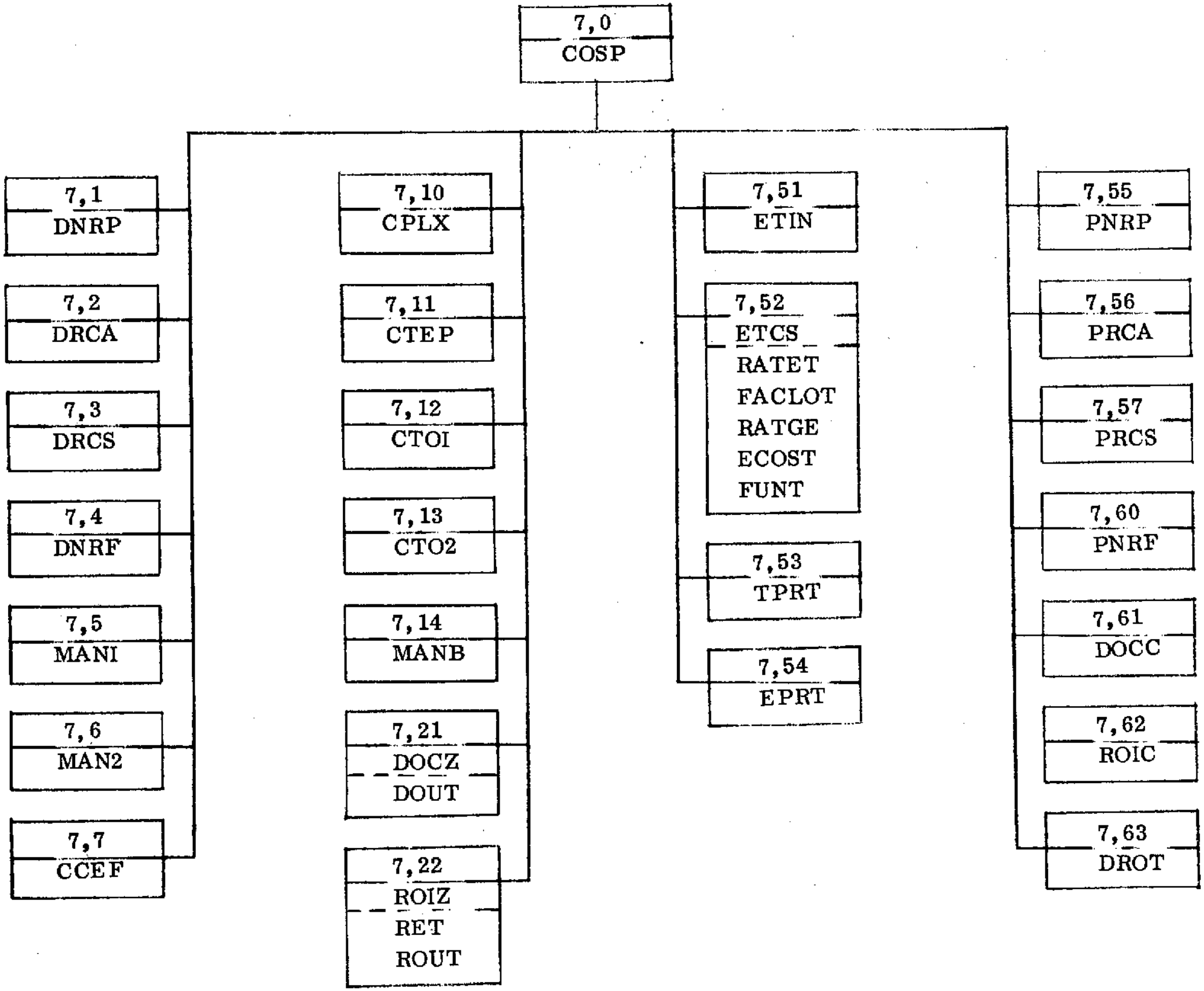


Figure 3-7 Overlay Structure for Cost Synthesis

OVERLAY 7,0 (COSP)

The COSP overlay serves as the overally driver for the manufacturing, tooling, engineering, total program and return-on-investment analysis, change and display program. This overlay also controls the main cost program control display menu.

OVERLAY 7,1 (DNRP)

The DNRP subroutine is a display routine for the non-recurring funded study parameters.

OVERLAY 7,2 (DRCA)

This subroutine displays the recurring RDT&E and investment cost for aircraft structural systems.

OVERLAY 7,3 (DRCS)

This subroutine displays the recurring RDT&E and investment cost for aircraft subsystems.

OVERLAY 7,4 (DNRF)

The DNRF subroutine displays the menu and change parameters for the total program cost analysis.

OVERLAY 7,5 (MANI)

The MANI subroutine displays the wing group manufacturing costs.

OVERLAY 7,6 (MANZ)

The MANZ subroutine displays the aircraft fuselage manufacturing cost.

OVERLAY 7,7 (CCEF)

This subroutine displays and controls the change parameters associated with the subsystem weight coefficients.

OVERLAY 7, 10 (CPLX)

This subroutine displays and controls the change parameters associated with the subsystem cost and the cost complexity factors.

OVERLAY 7,11 (CTEP)

This subroutine displays and controls the change parameters associated with the tooling and engineering cost analysis.

OVERLAY 7,12 and 7,13 (CTOI and CTOI)

These subroutines display and control the balance of cost change parameters not previously accounted for in overlay's 7,7 through 7,11.

OVERLAY 7,14 (MANB)

The MANB subroutine displays the tail group manufacturing costs.

OVERLAY 7,21 (DOCZ)

This subroutine is the driver and the analysis routine for direct operating costs. The subroutine resident within this overlay is DOUT which provides the direct operating cost output display.

OVERLAY 7,22 (ROIZ)

This subroutine is the driver and the analysis routine for return-on-investment. Subroutines that are resident within this overlay includes RET (establishes analysis year rates as a function of predetermined reference year), and ROUT (provides the return-on-investment output display).

OVERLAY 7,51 (ETIN)

The ETIN subroutine reads and stores the engineering and tooling cost input data and parameters.

OVERLAY 7,52 (ETCS)

This subroutine is the driver for the engineering and tooling cost analysis. Subroutines that are resident within this overlay includes RATET (establishes the tooling cost rate data), RATEE (establishes the engineering cost rate data), FACLOT (analysis for the aircraft lot sizes), ECOST (analysis for the engineering cost), and FUNT (analysis for the first unit costs).

OVERLAY 7,53 (TPRT)

This is a print subroutine for the tooling cost output.

OVERLAY 7,54 (EPRT)

This is a print subroutine for the engineering cost output.

OVERLAY 7,55 (PNRP)

This is a print subroutine for the non-recurring funded study costs.

OVERLAY 7,56 (PRCA)

This is a print subroutine for the recurring RDT&E and investment costs for aircraft structural systems.

OVERLAY 7,57 (PRCS)

This is a print subroutine for the recurring RDT&E and investment costs for aircraft subsystems.

OVERLAY 7,60 (PNRF)

This is a print subroutine for the total program costs.

OVERLAY 7,61 (DOCC)

This subroutine is the control for direct operating cost change parameters.

OVERLAY 7,62 (ROIC)

This subroutine is the control for return-on-investment change parameters.

OVERLAY 7,63 (DROT)

This subroutine is the output display for the return-on-investment analysis.

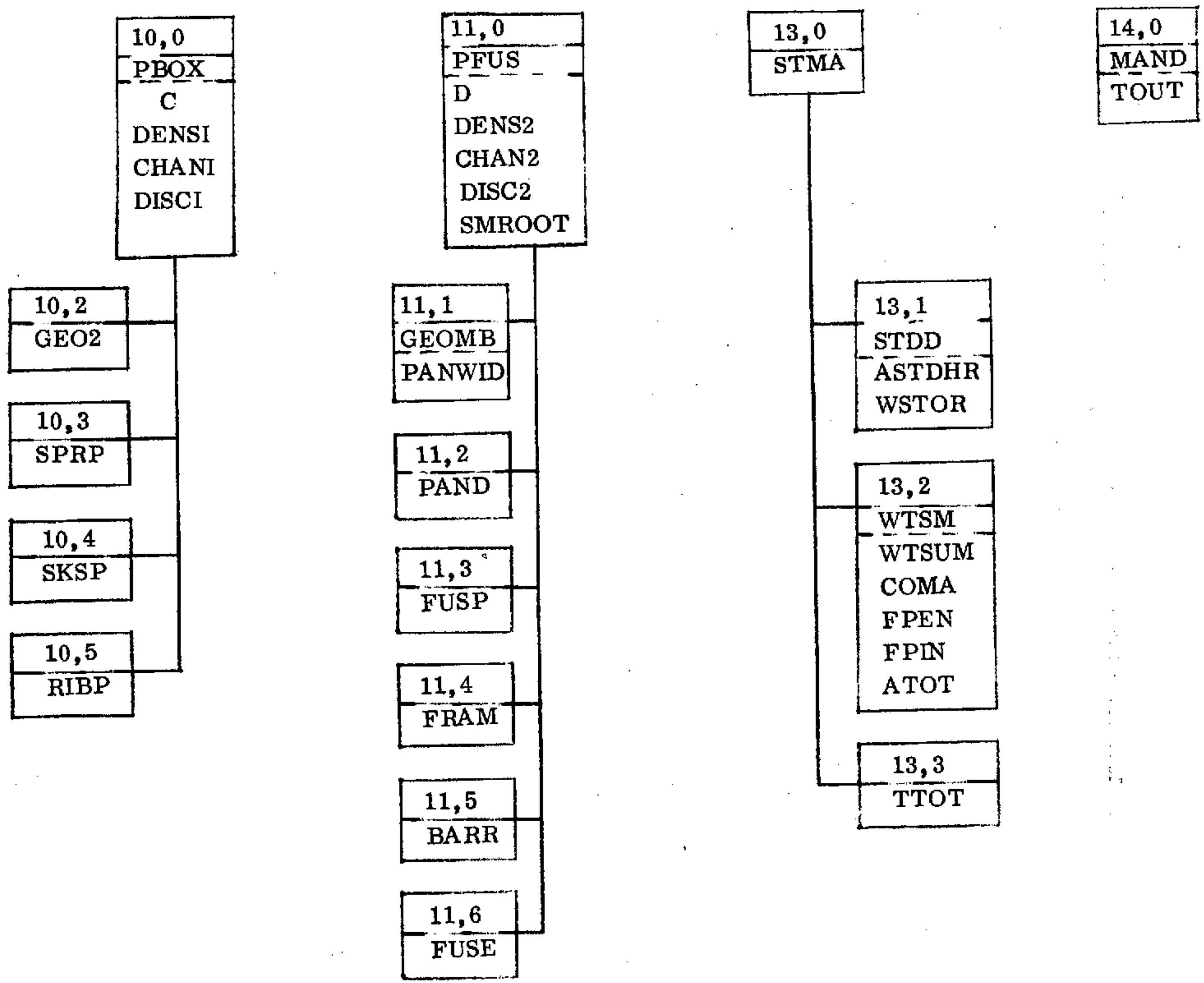


Figure 3-8 Overlay Structure for Part Definition

OVERLAY 10, 0 (PBOX)

PBOX is the driver portion of the aerodynamic surfact structural box part definition overlay. Subroutines resident within this overlay includes C (contains block data for input), DENSI (array of aerodynamics surface parts), CHANI (sets index to set material type or establish existing material for aerodynamic surface part), and DISCI (stores aerodynamic surface part data for retrieval and print).

OVERLAY 10, 2 (GEO2)

The GEO2 subroutine contains miscellaneous geometry information required by the part definition routines that are not developed within the aerosynamic surface structural box subroutines.

OVERLAY 10, 3 (SPRP)

The SPRP subroutine contains the part definition functional logic, and the actual and material weight equations for typical spar components including caps, webs, stiffeners, clips and fasterners.

OVERLAY 10, 4 (SKSP)

The SKSP subroutine contains the part definition functional logic and the actual and material weight equations for lifting surface skin panels.

OVERLAY 10, 5 (RIBP)

The RIBP subroutine contains the part definition functional logic and the actual and material weight equations for typical rib components including caps, braces, skins, and fasteners.

OVERLAY 11, 0 (PFUS)

PFUS is the driver portion of the fuselage shell part definition overlay. Subroutines resident within this overlay includes D (contains block data for input), DENSZ (array for fuselage shell parts), CHAN2 (sets index for material type or establish existing material for fuselage shell parts), DISC2 (stores fuselage shell parts data for retrieval or print), and SMROOT (provides square root calculations).

OVERLAY 11, 1 (GEOMB)

The GEOMB subroutine contains miscellaneous geometry intormation required by the part definition routines that are not developed within the fuselage shell structural

analysis program. The subroutine PANWID is also resident within this overlay and is used to define the fuselage panel width dimensions.

OVERLAY 11,2 (PAND)

This subroutine defines the fuselage barrel panel end points and overall dimensions.

OVERLAY 11,3 (FUSP)

Subroutine FUSP develops and defines the fuselage detail parts.

OVERLAY 11,4 (FRAM)

Subroutine FRAM develops and defines the fuselage frame detail parts.

OVERLAY 11,5 (BARR)

This subroutine accumulates the parts required to define complete fuselage barrel sections.

OVERLAY 11,6 (FUSE)

This subroutine accumulates the parts required to define the complete fuselage shell.

OVERLAY 13,0 (STMA)

The STMA subroutine is the primary driver for the standard hour and material cost analysis program.

OVERLAY 13, 1 (STDD)

The STDD subroutine is the driver for the standard hour analysis and part definition storing subroutines. Subroutines that are resident within this overlay includes ASTDHR (computes standard hours as a function of part and shop operation) and WSTOR (stores data on standard hours, parts and material for future use in the material cost analysis.).

OVERLAY 13,2 (WTSM)

The WTSM subroutine is the driver for the material cost analysis routines. Subroutines that are resident within this overlay includes WTSUM (retrieves standard hours, parts and material data from the WSTOR routine for use in the COMA routine), COMA (computes material cost for detail parts), FPIN (input data for the fuselage penalty analysis) FPEN (computes fuselage penalty weight data for doors, floors

and windows), and ATOT (totals detail parts data and categorizes information for printout).

OVERLAY 13,3 (TTOT)

The TTOT subroutine totals the weight and cost data and stores data in proper categories for summary and detail printout.

OVERLAY 14,0 (MANP)

The MANP subroutine is the driver for the manufacturing cost printout routine. Resident within this overlay is subroutine TOOT which calls data from TTOT and prints in summary and detail formats.

3.3 APPLICATIONS

The program is intended for use at the preliminary design level, and requires a minimum amount of input data. However, the actual depth of analysis reflected internally by the program allows its use to be extended to a degree into the detail design stage. To accomplish this purpose the program was designed to accept much of its input data on an optional basis. The so-called optional input is comprised of parameters that are not always available at the preliminary design stage, but that are often defined at a slightly later stage. These variables may be input by the user if they are known; or, in the absence of a direct input, values are computed internally by the program.

It is intended that the program be applied to the investigation of weight and cost sensitivities of airframe structures to various design alternatives. The advantage provided by the program is its ability to make cost and weight tradeoff studies at several levels of consideration. For example, weight and cost data can be related directly to key system parameters at the vehicle mission level such as payload, speed, range, and landing field requirements. At the vehicle configuration level, data can be related directly to surface areas, span, sweep, taper, etc., and fuselage length, slenderness, etc. At the major component level comparisons can be made between different materials, modes of construction, detail part makeup, etc. The program provides a means of refining aircraft design in terms of cost and weight to a high degree of detail.

The current version of the program is directed mainly at subsonic transport aircraft. Some factors limit consideration of other aircraft types. The RULES subroutine in the vehicle synthesis process derives a wing loading from the landing field length requirements. This is a typical design feature for transports but not necessarily one for high performance aircraft. This limitation may be circumvented by inputting directly a value for wing loading (or wing area) and fuel weight. In this case the RULES subroutine is bypassed.

The assumed structural arrangements and parts lists are those of a typical transport. Data from the DC-10 fuselage, 880 wing, C-14 empennage, and C-5 empennage were used to establish the data base for the program. There is no reason a variety of aircraft types could not be analyzed using the program if the parts lists and associated analyses were extended.

The program was designed to be run both in a batch mode or in an interactive graphics mode of operation. For the latter case, control of the program is transferred directly to the user. From the graphics console the user may study the output from various portions of the program, make changes to the input and recompute, plot the effects of changes in various vehicle parameters, check overall vehicle balance and plot the center of gravity envelope for a specified mission, tailor an area ruled fuselage shape, and inspect a three-view representation of the sized vehicle. The advantage of the interactive graphics interface is that the program user can work with the computer

in real time, combining the rapid response and data handling capabilities of the computer with human judgement and direction. In this way the user can direct the design analysis process step by step, immediately seeing the effect of any changes made.

It was anticipated that a program of this type would be updated and refined on a fairly continuous basis. For this reason the program incorporates a highly modularized format. Each subroutine was made to be as independent as possible of the rest of the program. Changes can be made to a single card or two, or an entire subroutine can be "unplugged" and replaced with a new one. The only requirement for the new subroutine is that the input/output interface be preserved for communication and data flow. Hence, it is possible to always have a usable version of the program available, even though new subroutines are in the process of being developed and checked out independently.

SECTION 4

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